SURVEY OF PROPULSION TECHNOLOGIES APPLICABLE TO CUBESATS

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ABSTRACT

At present, no Cubesat has flown in space featuring propulsion. This was acceptable as long as CubeSats were flown mostly as university experiments. As CubeSats become of interest to other users in the government and industry communities as well, a larger range of capabilities may be required than exhibited so far, while maintaining the uniqueness of the Cubesat platform. Propulsion capability is crucial in increasing mission capabilities of future CubeSats, such as orbit change and raising, formation flying, proximity operations, fine attitude control, or drag-make-up and de-orbit. While some of these tasks may be accomplished with propellantless devices, their applications are limited, applicable mostly to a single task, and bear their own risks. In this study, a survey was conducted of propulsion technologies applicable to CubeSats. Only few off-the-shelf design solutions exist today. The survey was thus expanded to such devices as well that are under significant development, and are approaching the required design envelope for CubeSats with respect to mass, volume, and power. In some cases, such as electric propulsion devices, CubeSat architectures themselves may need to be adapted, required to feature deployable solar arrays to increase power capabilities. Given the vast scope of this survey, only thruster technologies could be surveyed. However, valves and other feed system components, as well as their integration, are equally important, but have to be left to a future survey. Three major propulsion technology areas applicable to CubeSats emerged when conducting this review: (1) Existing technologies, such as butane systems, pulsed plasma thrusters, and vacuum arc thrusters are applicable to CubeSats today with no or only minor changes, (2) New thruster technologies under significant development, such as hydrazine monopropellant systems, ion engines, or colloid thrusters could be adapted to CubeSats with some further development, especially also in other subsystem areas such as feed systems and power processing units. They will also require increased power capabilities, and (3) emerging technologies, such as micro electrospray arrays and micro cavity discharge arrays that offer even higher flexibility due to scalability for CubeSats, and enable compact integration.

INTRODUCTION

CubeSat Background and the Case for CubeSat Propulsion

CubeSats are attaining increased intention by government and industry. Attracted primarily by their low cost, mass, and size, new mission paradigms may be explored using this spacecraft architecture, such as low cost technology demonstrators, formation flying missions consisting of swarms of these small spacecraft, or inspector satellites. CubeSats to date come in several standardized architecture sets. The original CubeSat configuration is a 10 x 10 x 10 cm cube approximately 1 kg in mass, termed “1U” (Fig. 1). “2U” and “3U” CubeSats are being offered today as well, consisting of 2 or 3 integrated “1U” units up to 10 x 10 x 30 cm in size and approximately 3 kg in mass [1], [2]. Nugent et al [1] and Venturini et al [2] list typical state-of-the-art CubeSat performances for 1- and 3-U sats. Some of the key CubeSat system performance parameters are repeated here in Table 1 [2]. As can be seen, besides tight mass and power constraints, power resources are particularly limited. This assumes no deployable arrays, as is currently the case for Cubesats. Bus voltages are significantly lower as the current spacecraft standard of 28V. Pointing accuracies at present also are fairly loose, limited by the availability of current attitude control actuators, mainly magnetic torquers on student-led CubeSat activities to date. In fact, to date university-provided CubeSats often do not feature any attitude control at all.

Figure 1: 1U CubeSat [1]

Table 1: Current and Projected CubeSat Capabilities (excerpted from Venturini et al. [2])

Parameter	Capability
Mass (kg)	1.0 (1U); 3.0 (3U)
Size (cm)	10 x 10 x 10 (1U); 30 x 10 x 10 (3U)
Power (W)	1.6 (1U); 10 (3U)
Payload Power (W)	0.5 (1U)
Pointing Accuracy (degree)	1
Pointing Control (degree)	0.5

The current capability constraints of Cubesats are rooted in their history and original intent, namely to provide students with a cost-efficient means to cheaply launch and operate microspacecraft in space and gain experience in actual, hardware oriented spacecraft design and operations. Developed in the 1990s as a result of a collaboration between Stanford and Cal Poly San Luis Obispo universities [2], the first launch of four university Cubesats took place in 2003 onboard a Russian Rockot vehicle and placed Stanford's QuakeSat-1 into orbit, among others [1]. Resource limitations, as typical for such student-led activities, limits subsystem capabilities that can be placed onto a CubeSat bus. Initial government led experiments using CubeSats, such as the Ames GeneSat-1, launched in 2006 aboard a Minotaur 1 [1], still exhibits this heritage in that in featured no attitude control system at all - due to the mission focus of a biological bacteria growth experiment, no such control was required.

As government and industry develop an increasing interest in CubeSats, required CubeSat capabilities are bound to become more demanding. The aforementioned CubeSat swarms, formation flying, inspector sats, or operations beyond Earth orbit etc in particular will all require substantial propulsive capability, either in terms of delta-v, impulse bit, specific impulse or thrust. Propulsion systems are generally complex, involving multiple components such as thrusters, tanks, valves, other flow component devices (regulators, pressure transducers, filters, etc) as well as power processing units for electric propulsion applications, valve drivers, etc, and involving potentially toxic or explosive propellants, all making the integration of propulsion systems into CubeSat architectures a relatively costly preposition, and so far typically beyond the budgetary capabilities of typical university led projects. Several attempts of developing micropropulsion capabilities, also in university environments, have taken place nonetheless in recent years, and will be reviewed in this paper. These were not exclusively focused on CubeSats, but involved the AF university nanosat activities, as well as various technology funding sources, both in the US and internationally [3], [4], [5], [6]. To date, no Cubesat has flown with onboard propulsion, and several of the university nanosat propulsion activities did not materialize into flight opportunities either. Thus, significant technology development in the micropropulsion field remains to be done, in particular in regards to actual flight applications.

Trends in CubeSat Propulsion Requirements

As for standard, larger scale conventional spacecraft, propulsion requirements will vary greatly depending on specific mission requirements. It this therefore impossible to try to accurately predict such requirements. However, a few general trends can be observed.

As pointed out in Table 1, mass, volume and power constraints by CubeSats dictate even more stringent constraints on any propulsion subsystem, given that it can only take up a portion of these resources. Large scale interplanetary spacecraft may have propellant mass fraction in excess of 60%, simple attitude control function require propellant mass fraction of but a few percent. Power resources available on CubeSats are likely the most constrained at the present time, not exceeding but a few Watts of total spacecraft power, however, also offer the largest growth potential. At present, CubeSats do not offer deployable array surfaces due to reasons of complexity and cost. Where these available, available power levels may be increased significantly, perhaps as high as 100 W for a 3U CubeSat [2]. Any survey of propulsion hardware applicable to CubeSats needs to account for the possibility of such "high-power" options.

Requirements for delta-v may vary widely depending on the mission. Orbit raising or plane change require approx. 135 m/s per degree of plane change and 0.6 m/s per km of altitude change in low-Earth orbit (LEO) at 250 km altitude according to Wertz and Larson [7] (Table 2), slightly decreasing with altitude. Depending on the altitude and plane change requirements, total delta-vs may range from a few hundred m/s to perhaps well in excess of 1 km/s. Depending on how much time is available to conduct such a maneuver, this may require either high-Isp electric propulsion options, or high thrust-to-power (T/P) low Isp or chemical propulsion options.

Table 2: Delta-v Requirements for Orbit raising and Plane Change at Different orbital Altitudes (excerpted from Wertz and Larson [7]

Earth Orbital Altitude	Plane Change (m/s)/deg	1 km Altitude Change (m/s)
250	135.35	0.58
500	132.86	0.55
750	130.97	0.52

Impulse bit (Ibit) requirements for attitude control have been estimated previously by Blandino, cited in Ref. [6]. Based on impulsive maneuvers, and a "deadband" time between firings of 20 and 100 seconds, respectively, required impulse bits for a 1 kg may be as low as 29 μNs for 1 degree (17 mrad) pointing requirements for 100 sec deadband intervals, and decrease for smaller pointing requirements of 0.02 mrad to as little as 0.034 μNs. Slew requirements, based on the arbitrary assumption to perform a 180 degree slew in one minute, may be as low as 0.06 mN [6]. These numbers may increase significantly, though, based on actual mission requirements. For example, a 10-kg microinspector satellite under previous development by JPL required up to 10 mN thrust to allow for sufficient maneuverability, including hazard avoidance margins [8]. In the latter case, a chemical butane system had to be chosen to operate the spacecraft within its available power levels of under 14 W, accounting for the draw on power resources by the other subsystems, while the aforementioned lower thrust values for slew may still be accomplished with highly scaled down high T/P electric propulsion systems, such as microfabricated colloid arrays.

Formation flying thrust requirements will depend highly on the needed positioning control for the formation, which in turn depends on the mission focus. Optical observations by means of interferometry have very tight positioning control and thrust requirements, ranging into the milli and even micro-Newton range even for larger scale spacecraft in the 1000 kg range, but may, given the physical constraints of CubeSats for the optics, not necessarily be an ideal candidate for CubeSat architectures. Looser requirements may result for distributed spacecraft architectures, whose sole purpose is to distribute subsystems over various small spacecraft to avoid detection, have redundancies in case of failures, etc. Given this large range of potential mission applications, thrust requirements may equally range widely. Micro-Newton to milli-Newton levels at a continuous or near-continuous thrust may be required.

Last but by no means least, propulsion subsystem component other than thrusters have to be considered. Although thrusters are naturally the first focal point of propulsion subsystem miniaturization, other components, such as valves, regulators, filters, will have to be miniaturized and tanks light-weighted to match CubeSat mass, volume and power envelopes. As will be seen in the course of this paper, even the smallest conventional commercial-off-the-shelf valve hardware, based on solenoid architectures, require several Watts to open and hold open, often exceeding current CubeSat power budgets. New ultra-low power technologies will need to be explored, such as piezovalves or micro-isolation valves [9]. Propulsion feed system architectures will also need to achieve higher degrees of integration to meet CubeSat volume constraints. Individual components plumbed and welded together will quickly use up valuable real estate even for the smallest available valve components. Integrated feed systems, based on MEMS [6] or ChEMS technologies (see Fig. 2) will be required. Multi-functional structure approaches, where thruster and valve components directly flange onto the tank are another option, and were previously explored at JPL under the micro-inspector project [8] (Fig. 3).

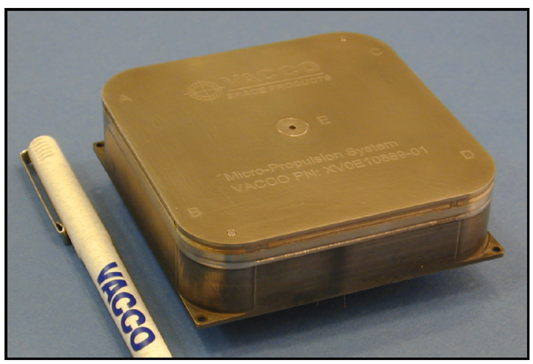


Fig. 2: VACCO Integrated ChEMS Thruster/Feed System Module

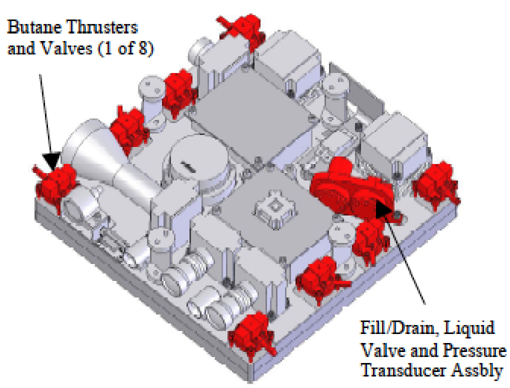


Fig. 3: Multi-functional Tank Structure with Integrated Propulsion Components for the JPL Micro-Inspector Satellite

Reviewing these requirements trends, it becomes quickly apparent that they are as varied as for conventional spacecraft and thus require, not surprisingly so, a suite of different propulsion systems ranging from low-power, high thrust chemical, via high T/P intermediate-Isp electric, and high Isp electric propulsion systems. Even propellant-less systems, such as sails or tethers may find applications in such cases, where de-orbit is required but no onboard propulsion system is required for the mission. However, in the case the CubeSat architecture features such a propulsion system, depending on required de-orbit time and available thrust and power, it may be more efficient to load additional propellant for that purpose.

Scope of this Survey

The large range in propulsion system requirements alluded to in the previous sections will be reflected in the survey of propulsion hardware undertaken in this paper, including chemical, electric and propellantless propulsion systems. This survey builds upon a detailed micropropulsion surveys conducted previously [6], [9], [10], with a substantial amount of new material having been included in this survey. Compared to a detailed survey conducted by one of the authors a decade ago [6] the increased level of international activities is particularly noteworthy, with US activities mostly focuses on DoD sponsored projects due to the demise of NASA's technology budget in recent years.

This survey will not be limited to off-the-shelf hardware exactly matching CubeSat requirements today. There are only very few thruster technologies out there that would fit into this category. Also, as mentioned, CubeSat system requirements may change in the future due to e.g. deployable solar arrays. Therefore, thruster technologies, either existent or under significant development, are being reviewed that either fit or approach the CubeSat envelope, thus pointing to areas of possible development where a match with CubeSat requirements may be met in the future.

The vast scope of this survey does not allow for the inclusion of valve and other feed system components, although their miniaturization and integration are at as important as similar efforts underway for thrusters. At this point the author may only refer to a previous survey of the vast amount of activities in this field, mostly confined to research activities [9]. Off-the-shelf miniature valve technologies can also be found in the cold gas thruster section of this paper. A more detailed survey of this field will need to be left to a future publication.

CHEMICAL PROPULSION OPTIONS

Monopropellant Thrusters – Hydrazine

Hydrazine thrusters are the workhorse for attitude control and small delta-v applications in conventional spacecraft applications. Hydrazine is decomposed catalytically in a catalyst bed consisting of iridium coated alumina pellets. Hydrazine thrusters emerged as early as 1949 through development work at JPL, and have been in flight applications since 1966 [6]. Commercially available thruster hardware, however, is too heavy and large, and thruster valves are too power consuming to be used on CubeSats even when used as a single main engine for delta v applications [6]. A recent development of a hydrazine milli-Newton thruster (HmNT) at JPL, however, may at least fit the mass and size envelope of a CubeSat for such an application [11], [12] (Fig. 4). While too large to be used in sets of multiple units for attitude control, one may contemplate the use of a single such thruster for small to intermediate delta-v applications. The use of hydrazine poses hazards in particular in the university environments due to its toxicity and flammability, however, future government or industry applications of CubeSats may not be as affected by such considerations die to the extensive experience in handling hydrazine in these places.

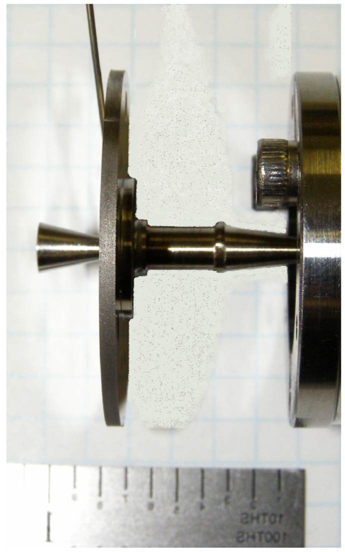


Figure 4: JPL Hydrazine Milli-Newton Thrusters (HmNT) by Parker et al [11], [12]

The HmNT was not developed, however, with CubeSat applications in mind, and rather targets precision pointing and formation flying missions of standard large scale spacecraft buses, allowing for the replacement of reaction wheels, eliminating wheel jitter and reducing mass if tied into an existing coarse hydrazine attitude control system. Typical HmNT characteristics are shown in Table 3. Noteworthy is the significant 100-fold reduction of the minimum impulse bit over state-of-the art COTS hardware to 50 μNs. Thrust levels have been measured on a JPL micro-Newton thrust stand to 129 mN. The thruster weighs 40 g in its current laboratory model stage, and is about an order of magnitude smaller in volume than SoA hardware. Valve power, however, remains too high for CubeSat applications with current power capacities (Table 1) at 8 W valve opening and 2 W valve holding power. However, small impulse bit performances, requiring fast acting valves will not be needed in a delta-v application, and outfitted with a slower acting, lower power valve it may serve this function on CubeSats, in particular if higher power versions become available in the future. Already, the HmNT would be suitable for attitude control on somewhat larger microspacecraft, in the 20-30 kg range.

Table 3: Small Hydrazine Thruster Performance Comparisons

Parameter	SoA MR-103H	HmNT [11], [12]	MEMS/Yuan and Li [14]
Thrust (mN)	1,070	129	0.74 – 1.44
Isp (sec)	220	150 (est.)	162
Volume (cc)	94	8	-
Power – Valve (W)	5 (open)/1 (hold)	8 (open)/1 (hold)	-
Power – Catbed (W)	1.5	0.25 (est.)	-
Min. Impulse (μNs)	5000	50	-
Mass (g)	195	40	-

Additional past developments include a lower-power miniature hydrazine thruster version envisioned by Platt [13], with Lee microvalves, operating at power levels of 0.5 W (holding), however, limiting the flow to 10^{-4} gram/sec and currently constrained at 120 psi pressure. MEMS based thruster versions were tested by Yuan and Li [14] in Taiwan. These thrusters were fabricated out of silicon using MEMS micromachining techniques. It featured a catalyst bed unit consisting of micromachined iridium coated pillars, and integrated planar MEMS micro-nozzle with a throat width of 60 μm and height of 77 μm, half angle of 22.5° and an expansion ratio of 15 (Fig. 5). An integrated electrostatically actuated valve was also provided as part of the assembly. Actuation forces of this type of valve are small [9], and the valve diaphragm was pre-stressed to increase contact forces with the seat, further aided by thermally stressing it through heat soakback from the thruster – a feature that can obviously not be exploited when the thruster has not been in use for extended periods of time. Thrust stand tests were performed and indicated 0.74 – 1.44 mN of thrust at 0.48 – 0.96 mg/sec of hydrazine flow (Table 3). Vacuum specific impulse was estimated at 162 sec at 0.99 mN thrust [14]. Future development could render such a thruster as an option for CubeSat attitude control, ideally in connection with HmNT-derived or similar devices for delta-v applications.

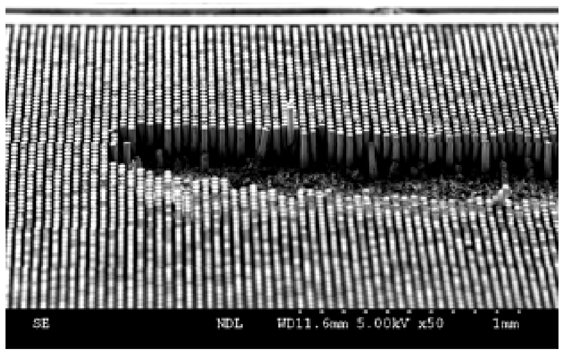


Figure 5: Micromachined Hydrazine Catalyst Bed featuring Ir-coated Si micromachined pillars [14]

Monopropellant Thrusters – Hydrogen Peroxide

Scharlemann et al. [15], [16] at Austrian Research Centres Seibersdoorf (ARCS) have been developing hydrogen peroxide micro-thrusters for microspacecraft applications (Fig.6). As for the HmNT described above, these thrusters are too large to be used for attitude control functions onboard a CubeSat, but maybe used as main engines for small to intermediate delta-v. Hydrogen peroxide has been used since the dawn of the space age in many famous, and some infamous roles, such as a propellant to power the fuel pumps of the V-2, and post-war US Redstone, Jupiter and Viking missiles, and as monopropellant on the early crewed US flights of Mercury, as well as the X-1 and X-15 rocket planes [6]. Its use in spacecraft applications decreased with the advent of higher performing hydrazine propellant.

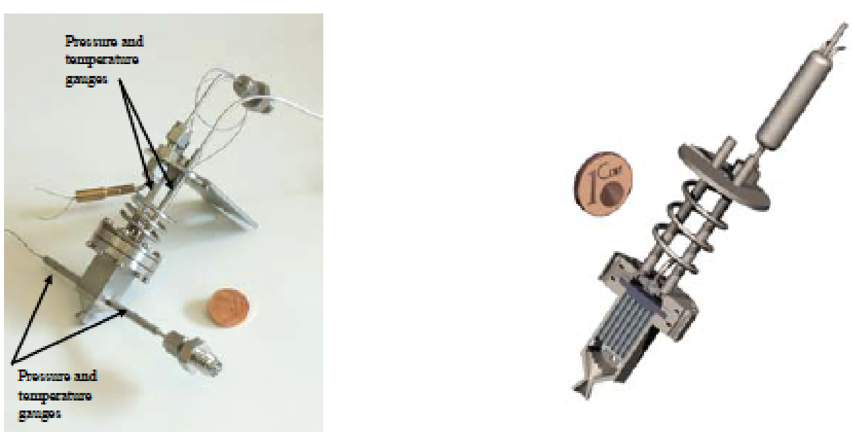


Figure 6: Miniature Hydrogen Peroxide Thruster by Scharlemann et al. [15]

However, the use of hydrogen peroxide has regained attention recently in the microspacecraft community, as its lower degree of toxicity may make it more amenable to use by university groups, which have led microspacecraft activities so far [6]. Measured thruster performances by Scharlemann et al [15], [16] indicate thrust capabilities in their laboratory prototype of 100 – 800 mN and a vacuum equivalent specific impulse of 153 sec (ambient specific impulse was determined to be around 100 sec). The thruster uses a monolithic catalyst bed impregnated with sodium permanganate [16] (Table 4).

Hydrogen-peroxide thruster development in the past was also underway by Sahara et al. in Japan [17] and by and subsequently by Kuan et al. [18], [19] in Taiwan, both similar in size as the Austrian device. No thrust and Isp performance are available to date in the literature for the Japanese device [17]. Kuan et al [18] paid particular attention to the thermal design of the catalyst chamber, and arrived at a device approx. 8 (dia.)x 13 mm in size with a throat diameter of 0.5 mm dia. and 5.8 g in weight, delivering 182 mN thrust at 101 sec specific impulse at ambient conditions, which was estimated to equal 221 mN of thrust and 125 sec of specific impulse under vacuum conditions [18] (Table 4). Silver flakes were packed into a catalyst bed retained by a stainless steel mesh. Typical operations requires preheating the bed with 10-15 Watts of power to attain steady state thrust values within 50 ms. Operations without pre-heating was conducted in laboratory tests and resulted in achieving steady state chamber pressures and thrust after 15 sec [18]. While this may be acceptable for delta-v applications, cat bed lifetimes may be compromised by cold starts, as for conventional hydrazine thrusters.

Table 4: Small Hydrogen Peroxide Thruster Performance Comparisons

Parameter	ARC Seibersdorf [15]	Kuan et al. [18]
Thrust (mN)	100-800	221
Isp (sec)	153	125
Size (mm)	-	8 x 13
Mass (g)	-	5.8
Heater Power (W)	-	10-15 (cat bed pre-heat)

MEMS based devices have been developed initially at NASA Goddard Space Flight Center (GSFC) by Hitt et al. [19] Hitt et al.'s [20] thruster consisted of an entirely MEMS fabricated planar catalyst bed and nozzle design machined into silicon. The catalyst consisted of micromachined silicon pillars, like those later reproduced by Yuan et al for the MEMS hydrazine thruster [14], coated in silver as the catalyst for peroxide decomposition (Fig. 7). The initial experiments were plagued by incomplete hydrogen peroxide decomposition, occurring only near the silver coated pillar surfaces, and not in the bulk of the propellant flow. This may have been due to incomplete mixing in the low Re number flow [20] or due to the large surface-to-volume ratio design of a highly thermally conductive Si chamber, which may have led to excessive heat losses [18].

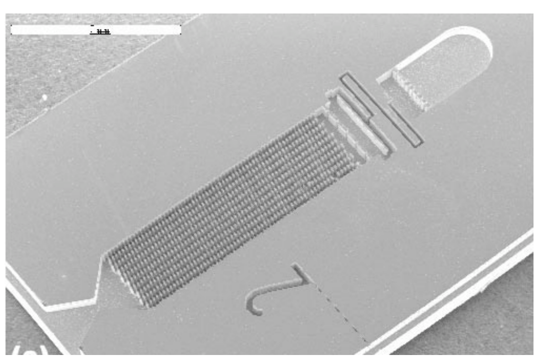


Figure 7: Micromachined Hydrogen Peroxide Thruster by Hitt et al [20]

Hydrogen peroxide thrusters may fill a similar function as HmNT derived thrusters for CubeSats, and provide small to intermediate delta v. The benefit of non-toxic hydrogen peroxide, however, must be weighed against storability concerns of this propellant. In a previous review [6] it was pointed out that hydrogen peroxide faces storability concerns. Hydrogen peroxide slowly decomposes when heated or exposed to a catalyst. Almost any organic residue can act as such a catalyst. Slow decomposition in propellant tanks can lead to tank overpressurization over time. However, Scharlemann et al. [16] point out that decomposition rates can be reduced by using high purity hydrogen peroxide, adding stabilizers, and surface passivations.

Cold Gas Thrusters

Cold gas systems offer the greatest degree of simplicity of all propulsion systems. Basically, gas from a high-pressure gas supply tank is vented through a valve and nozzle to produce thrust. Cold gas thruster, besides their simplicity, offer low impulse bits and contamination-free and non-toxic propellant options, and are space-proven technology since the 1960's [6]. They are particularly suitable for space applications where a combination of low impulse bit, exposure to contamination sensitive spacecraft surfaces, and low total impulse requirements is given. Given that they were designed for low-impulse bit attitude control for spacecraft, existing thruster hardware is small and light weight. Table 5 shows SoA cold gas thruster hardware. As can be seen, thrust values range from 4.5 mN to 100 N. The smallest thrusters, such as Moog's thrusters originally developed for attitude control purposes for a non-realized Pluto Fast Flyby mission, weigh as little as 9 grams and are as small as 12 x 35 mm.

However, most of these thrusters were never designed with microspacecraft applications in mind, and the power levels required for valve actuation, although small by conventional standards around 10 W (opening) down to < 1 W (holding) for the smallest units still far exceed today's CubeSat capabilities (see Table 1), in particular if one further takes into account that for attitude control purposes at least two thrusters are typically fired in common. One exception is the thruster developed by Marotta, which was envisioned for microspacecraft use on the NASA's ST-5 New Millennium mission, and has a pull-in power of less than 1 W, however, at the expense of a larger thruster mass of around 70 grams.

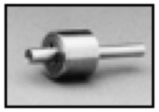
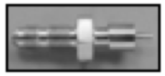
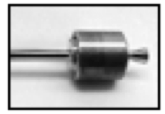
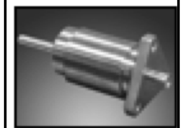
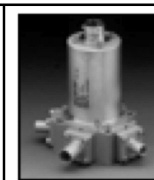
Other than the aforementioned miniature hydrazine and hydrogen peroxide thrusters, cold gas thrusters are also not suitable for delta-v applications since specific impulses are low, typically around 65 sec for nitrogen propellant, leading to large propellant mass fractions. This increased propellant mass fraction further increases propellant system mass since heavy weight tankage is required to contain the high-pressure gas supply. The large storage pressure also creates a risk of propellant leakage across valve seats, in particular after valves have seen extended use throughout the mission, increasing the likelihood of deposition of contaminants onto valve seats [6]. Finally, the cold gas thrusters listed in Table 5 require conventional integration using tube-welding approaches. This leads to relatively bulky systems in volume constrained CubeSat systems.

MEMS-based cold gas thruster systems have been developed in Sweden by Stenmark et al. [4], [6], [21] – [23]. These thrusters eliminate the integration concerns mentioned above by combining four thrusters with integrated valves, featuring stacked piezoactuators, filters and heaters (Fig.8). The use of piezovalves also reduces power requirements, and stacked piezoactuators provide sufficient valve stroke at merely 24 V [6], [22]. Despite these impressive miniaturizations and integration schemes, this system will still have to content with high pressure gaseous propellant storage and the aforementioned concerns resulting from it.

A novel gas generator concept has been under development in the Netherlands, addressing these high-pressure gaseous propellant storage and leakage concerns of conventional cold gas systems [24]. Rather than relying on high-pressure tanks, propellant is stored in solid gas generator cartridges and upon ignition of these released into a plenum from which nitrogen gas can be drawn for propulsive use. Since only a fraction of the gas is only stored in the plenum shortly before use at any time of the mission, while the bulk of the nitrogen gas is bound in the solid gas generator cartridges, tank volume requirements and leakage concerns are reduced. Originally developed by TNO and Bradford Engineering

Corp. in the Netherlands, the cartridges release pure nitrogen from a proprietary substance upon pyrotechnic ignition (Fig.9). Since pyrotechnic igniters are banned on CubeSats, Rackemann et al. [24] developed a resistive igniter, estimated to require approximately 2.5 W for 30 seconds [24].

Table 5: Cold Gas Thruster Performance Comparisons

							
Manufacturer	Moog	Moog	Moog	Moog	Marotta	Moog	Moog
Model	58X125A	58E143 58E144 58E145 58E146	58E142	58E151	Cold Gas Micro-Thruster	58-118	50-820
Propellant	N2	N2	N2	N2	N2	N2	N2
Thrust (N)	0.0044	0.016 - 0.040	0.12	0.12	0.05 - 2.36	3.6	52-105
Mass (g)	9	40	16	70	<70	23	430
Size (mm)	11.9x34.7	13.97x57.2	14x20.3	19.05x 40.87		6.6x25.4	98.2x104.1
Valve Power (Open) (W)	10	10	<35	10.5	<1	30	47
Valve Power (Hold) (W)		1					
Isp (sec)	65	>60	>57	65	65	65	65
Operat. Pressure (psia)	0-50	0-36	50-300	0-400	100-2240	230	215-2515
Proof Pressure (psia)	300	290	600	1015	3360	1115	3765
Burst Pressure (psia)		508	1200	1615	5600		6265
Response (Open) (ms)	2.5	2.5	3.5	5	5	<4	<10
Response (Close) (ms)	2.5	2.5	3.5	3	5		<10
Minimum Ibit (mNs)					<44		
Life (No. of Cycles)	>15,000	500,000 - 2,000,000	20,000	1,000,000		>10,000	>6,000
Status	Flight Qual	Flight Qual	Flight Qual	Flight Qual	Flight Qual	Flight Qual	Flight Qual
Comments	Brlliant Pebbles, SAFER, Pluto Fast Flyby	CHAMP, GRACE	SIRTF	SIRTF	Developped for GSFC Nanosats, ST-5	SCIT, SAFER, Pluto Fast Fly-by	COMET, Pegasus Cluster of 3 thrusters: 2 @ 52N, 1 @ 105N

(a) (b)

Figure 8 (a) MEMS –based Cold Gas Thruster Module, (b) Cold Gas Thruster Chip [4]

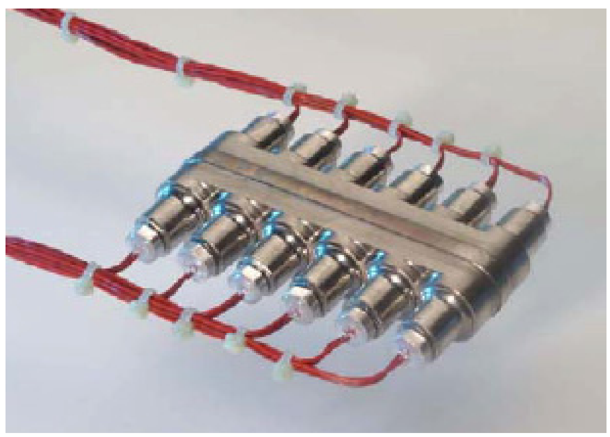


Figure 9: Gas Generator Cartridges Demonstrator [24]

Miniaturization of the demonstrator unit is required as well, requiring smaller cartridges to be used in sequence to allow for a more consistent gas supply from a smaller plenum as required for Cubesats. Rackemann et al estimates that volume requirements of a gas generator system may be reduced by a factor of six over a high-pressure tank based cold gas system, however, mass reduction is significantly less, around 20%, due to cartridge mass, its solid base material, and required plumbing.

Liquefied Gas Thrusters

Liquefied gas thrusters use propellants, such as butane or ammonia, which can be stored in their liquid form, yet phase transfer into gas upon expansion. Within the propellant tank, due to self-pressurization due to ambient heating from the spacecraft, the gas remains in its liquid form for the largest part, the ullage being filled with gas-phase propellant at its vapor pressure for the given tank temperature. For butane, this pressure value ranges between 40 – 100 psia for typical spacecraft temperatures. This allows much lighter, lower pressure tank constructions, significantly reduces the leakage concerns, and yet maintains the simplicity of cold gas thruster technology upon expansion of the liquid propellant from the tank into a plenum.

Butane thrusters were first build and flown onboard the SNAP-1A spacecraft by Surrey Space Centre in England in 2000 [25]. The Surrey system pursued a low-cost approach by constructing the tank from bend large diameter tubing, capped off at the ends with fittings. Given the low pressure propellant storage, this was feasible. Unfortunately, the Surrey system did not use a plenum tank. Butane was expanded from the tank directly into the thrusters. Incomplete vaporization of butane may have led to the expulsion of butane in its liquid phase, which leads to extremely low specific impulses and rapid propellant expenditure [25].

A JPL micro-inspector concept employed such a plenum [8]. The JPL micro-inspector was based on a pre-cursor, the Low Cost Adjunct Microspacecraft (LCAM) pioneered by Collins at JPL [8]. Here, the central butane liquid propellant tank was surrounded by a plenum, into which the butane was allowed to expand and vaporize completely. This was aided by waste heat from the spacecraft transferred into the tank by mounting heat-generating avionics flush with the tank. This “multi-functional tank” (MFT) approach, invented by Collins, is seen in a later iteration in Figs 3 and 10 for the Micro-Inspector spacecraft. As can be seen, the plenum surrounds the liquid tank, including its weld and so acts as an additional safety feature containing the inner liquid tank. The low tank pressure (less than 100 psi even in worst temperature cases) allowed for a flat tank geometry except for slightly curved internal tank surfaces to facilitate the desired maximum tank pressure. The flat surfaces provided interfaces for avionics board mounting (providing heat of vaporization to the butane propellant and in turn allowing for cooling of the avionics), as well as flanged thruster and valve components (shown in red in Fig. 3). A latching and liquid control valve allows transfer from the liquid into the plenum tank based on pressure sensor feedback. From the plenum tank, eight thrusters are fed to produce attitude control and small delta-v for Micro-Inspector. Total delta-v capability was 15 m/s, sufficient for the intended use of spacecraft inspection of the host spacecraft.

As for CubeSats, Micro-Inspector was power limited to 10 W, and lower power valve technology was required. VACCO Industries developed the cold gas thrusters featuring piezovalve technology, as well as the piezovalve based liquid control valve, miniature latch valve and pressure transducer assembly, as well as the miniature fill 7 drain valve. The piezovalve allowed for opening operations of 1 W or less based on ramp up speeds, and micro-Watt levels of holding power, and is thus uniquely suited for microspacecratt applications. The cold gas thruster provide up to 25 mN of thrust and the specific impulse was estimated to be around 70 sec based on thrust and mass flow measurements conducted at JPL (Fig. 11). Thruster performance data are summarized in Table 6.

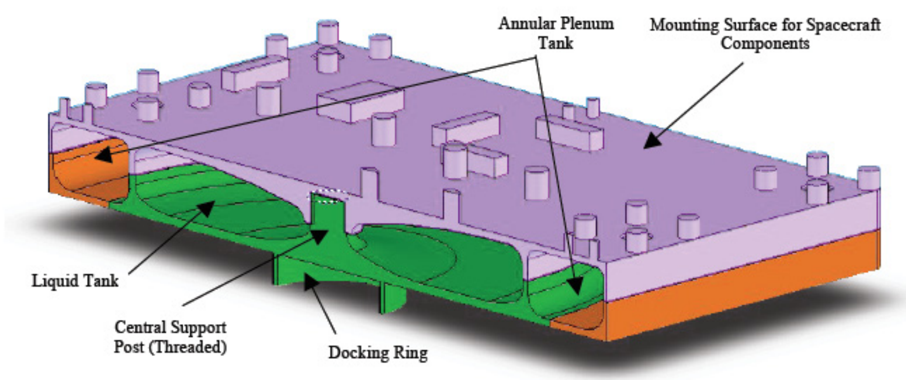
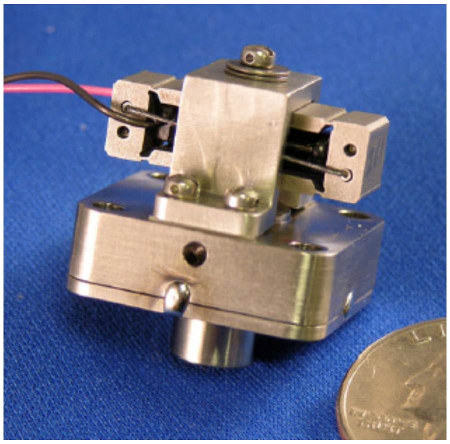
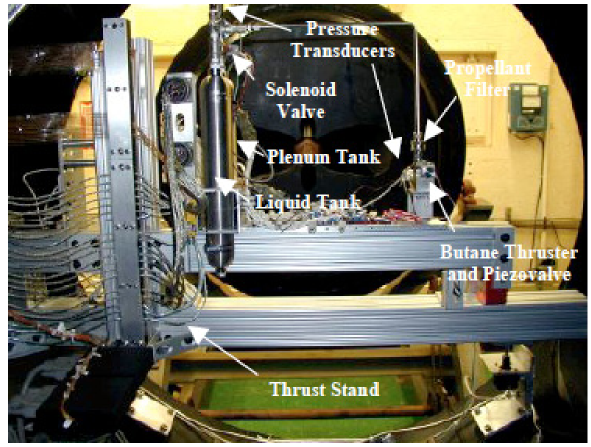


Fig. 10: JPL Micro-Inspector Multifunctional Tank Design featuring Liquid and Plenum Tank [8].





(a) (b)

Fig. 11: (a) VACCO Piezoactuated Butane Thruster [27] (b) on JPL Micro-Newton Thrust Stand [8]

Table 6: Micro Butane Thruster Performances

Vendor	Propellant	Thrust (mN)	Isp (sec)	Mass (g)	Power (W)
VACCO	Butane	10-25	70	30	1 (open)/ 10 μW (hold)

In parallel to the aforementioned activities, VACCO developed an integrated butane propulsion system based on its ChEMS™ (Chemical Etched Micro Systems) technology [26] – [28]. ChEMS™ is a VACCO developed metal-micromachining technique. While not allowing for the same level of precision micro-features to be machined as by MEMS into silicon, it far exceeds traditional metal-machining techniques and is sufficient for the types of miniaturizations needed for CubeSats. It relies on chemically etching metal disks that then are diffusion bonded into stacks to build up 3-D structures. Dropping in micro-coils etc allows complex manifolds with integrated valves, thrusters, etc to be machined [28].

Developed originally for the MEMS PicoSat Inspector (MEPSI) by Aerospace Corporation, this system was the first butane propulsion system designed specifically for CubeSats. Termed the Micro Propulsion System (MiPS), it consists of a self-contained unit featuring five thrusters, integrated liquid and plenum tank with internal control/isolation valve, pressure and temperature sensors, and filters (Fig. 12). The self-contained unit thus contains all propulsion components and the butane fuel, allowing for an elegant modular design that can be cleanly interface with a Cubesat bus by merely attaching it on to its side (Fig. 13). In fact, in addition to the micro-pulsed plasma thruster (PPT) by Busek and the Vacuum Arc Thruster (VAT) by Almeda Applied Science Corp (AASC Inc.) to be reviewed in the Electric Propulsion Section further below, it is one of only a few propulsion systems that are available off-the-shelf today, ready for CubeSat use, and able to operate within its current spacecraft mass, volume, power, and bus voltage constraints. As the PPT and VAT are very low thrust and inherently impulsive thrusters, the VACCO ChEMS™ butane module is the only readily available chemical thruster system, able to operate with continuous thrust force.

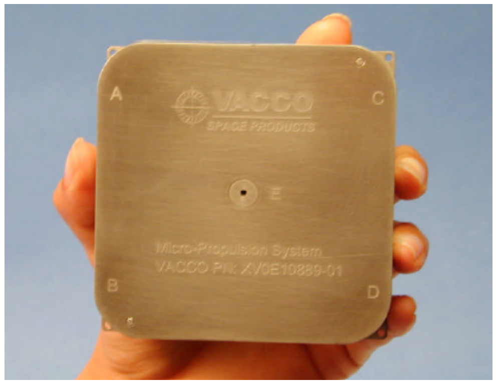


Fig. 12: VACCO MiPS Module [27]

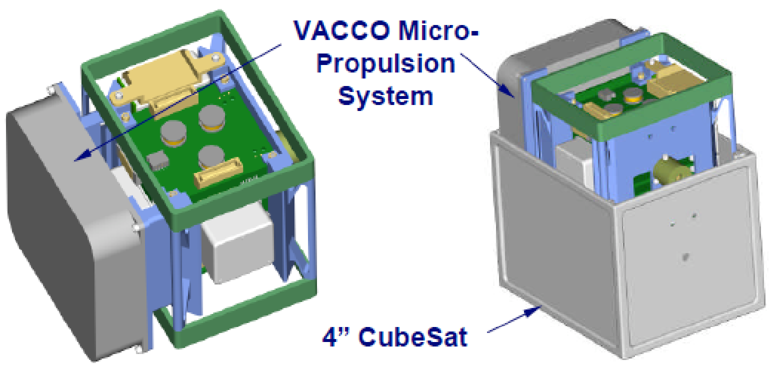


Fig. 13: VACCO MiPS CubeSat Integration [28]

Typical performance data for the VACCO MiPS is listed in Table 7. Total system dry mass (and therefore including the tank) is 456 grams, propellant capacity is 53 g of isobutene. The thrusters deliver 55 mN thrust, and the total impulse capability of the units as shown in Fig. 12 is 34 Ns at a total delta-v capability of 34 m/s. The valves are of the solenoid type and have demonstrated over 90,000 cycles, demonstrating its design maturity.

VACCO extended the MiPS design in collaboration with Boeing [27] to the “Palomar” design (Fig. 14) . This system has increased capability, and in addition to butane may be operated on nitrous oxide (N_2O). Nitrous oxide was found to decompose exothermally when heated. The Palomar system features two decomposing thrusters for delta-v applications, and 16 cold gas N_2O thrusters for attitude control, plus tankage, control valves, filters, pressure and temperature sensors. It weighs 1.39 kg dry, has a propellant capacity of 96 grams at 264 – 1500 psia storage pressures for nitrous oxide, requiring spheroid tank shapes compared to lower pressure butane applications, which can employ flat tank designs. Performances estimated based on nitrous oxide propellant are 141 Ns (decomposing) and 56 Ns (cold gas). The Palomar system has also been operated on isobutane [28].

These demonstrated performances and high levels of integration aided by low pressure butane storage make butane systems ideal candidates for low delta-v applications on CubeSats and attitude control, where rapid reaction and slews are required and power levels are limited.

Fig. 14: VACCO/Boeing Palomar MiPS Module [27]

Table 7: VACCO MiPS and Palomar Module Perfornances [27] – [28]

Parameter	VACCO MiPS	VACCO Palomar
Propellant	Butane	N_2O
Thrust (mN)	55 (per thruster)	-
Mass (g)	456	1,393
Propellant Mass (g)	53	96
Total Impulse (N-s)	34	56 (cold gas); 141 (decomposing)
Delta –v (m/s)	34 (for 1 kg MEPSI CubeSat)	-

Solid Rocket Motors (SRMs)

Solid rocket motors have been used since the dawn of the space age, mostly in the function of kick-stages or orbit insertion [6}. It is for these functions that they may become of interest as well for CubeSat based missions, i.e. orbit raising, trajectory insertions, rapid de-orbiting etc. As for conventional spacecraft, SRMs would not be integrated into the spacecraft bus, but the motor will be attached to the spacecraft, in this case the CubeSat. Thus, the SRMs do not necessarily have to meet CubeSat mass or volume constraints, although there will be some short term power demands for motor ignition. However, for spacecraft as small as CubeSats, thrust levels and resulting vehicle accelerations are a major selection criteria.

Most SRMs today use a propellant consisting of an HTBP (Hydroxyl-terminated Polybutane) matrix with aluminum particulates as fuel additive and ammonium perchlorate (NH_4ClO_4) as oxidizer. Since the propellant is solid, and now valves are required, SRMs allow for tight packaging and easy integration for sizable delta-v requirements given that specific impulse values even for small motors reach above 250 sec. Disadvantages are that they are generally not restartable and require orbit trimming due to thrust and total impulse uncertainties, thus general requiring additional propulsive capability unless inaccuracies of the solid burn can be accepted. In addition, as mentioned above, the selection of suitable small motors is limited. Most smaller motors have been used as retro rockets, or for stage separation on launchers and were required to deliver large thrust values over very short burn durations often under 1 sec [6]. Resulting vehicle accelerations for a CubeSat would be excessive, requiring costly requalification of spacecraft subsystems for high accelerations.

A few exceptions to this trend are listed in Table 8. The STAR 5A motor by ATK is a so called end-burner, burning slower than most motors, producing 169 N of thrust over a burn duration of 32 sec at an Isp of 250 sec (Fig. 15). Thrust forces and accelerations would still be high, but significantly less than for most space qualified off-the-shelf SRM design [6]. The STAR 5A could provide around 1.3 km/s delta-v to a 1U CubeSat at an average vehicle acceleration of around 4 g. The ATK STAR 4G listed in Table 8 was designed for the New Millennium ST-5 mission (Fig. 16). At 257 N thrust (average) and a burn time of just over 10 sec, accelerations would be even larger. A 1U CubeSat could attain a delta-v of approximately 1.35 km/s at an average vehicle acceleration of around 13 g, in large part due to the shorter burn time.

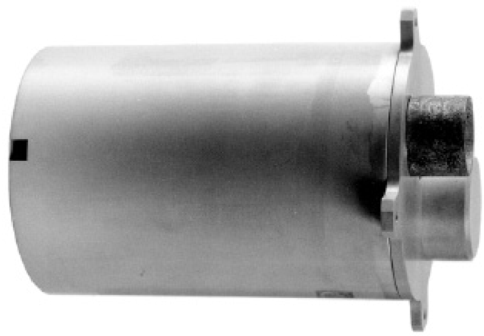


Fig. 15: ATK STAR 5A Solid Rocket Motor (5 in. dia.) (from ATK STAR Motor Catalogue)

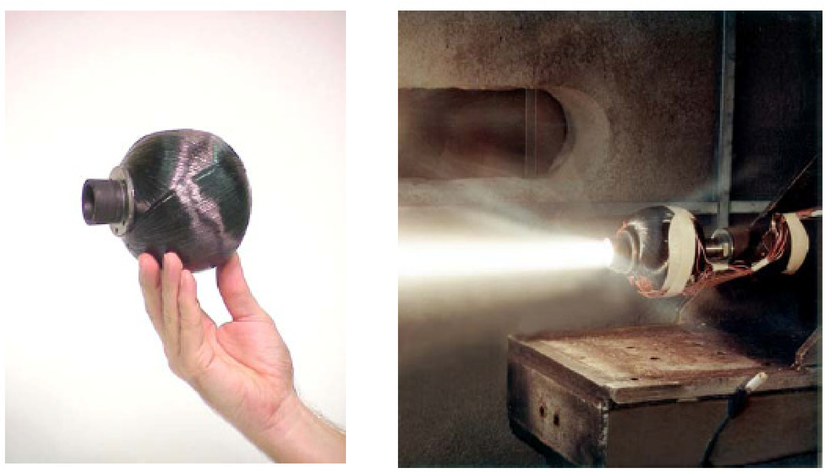
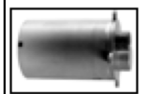


Fig. 16: ATK STAR 4G Solid Rocket Motor (from ATK STAR Motor Catalogue)

An even smaller motor was designed by Teasdale et al. [29] for the Smart Dust mission concept, where distributed sensor arrays placed on microspacecraft to be dispersed in a region of space. The motor was designed to facilitate the initial dispersion of these sensor arrays, not formation control subsequent to that. The motor consisted of an Alumina ceramic thrust chamber, and silicon micromachined nozzles with integrated polysilicon heaters for igniters. The igniter require approx. 0.2 – 0.3 W for less than 1.5 sec. HTPB fuel and ammonium perchlorate oxidizer propellant combination was used. Interestingly, no Al particles were added to the propellant mix as Al slag formation during burning clogged the 0.5 mm dia. micro-nozzle throat. The motor weighed 0.75 gram fueled with a propellant mass of 0.12 gram. Combustion chamber length is 12.7 mm, OD is 4.8 mm and ID is 3.2 mm. Peak thrust values of 15 mN were measured over the course of the first 1 sec of the burn, settling into a steady state thrust value of 2 mN over a burn time of approx. 7 sec. The specific impulse was given as only 14 sec

[29]. From that a total impulse value of approx. 16.5 Ns can be estimated, using the propellant mass above.

Table 8: ATK Small STAR Solid Rocket Motor Performances

		
Manufacturer	ATK	ATK
Model	STAR 4G	STAR 5A
Propellant	TP-H-3399	TP-H-3399
Thrust (avg) (lbf)	58	38
Thrust (max) (lbf)	69	38
Mass (Loaded) (lbm)	3.3	10.24
Propellant Mass (lbm)	2.16	5.05
Burnout Mass (lbm)	1.07	5.08
Case Material	Graphite-Epoxy	Al
Size (in)	4.45x5.43	5.13x8.84
Burn/Action Time (sec)	10.3/10.8	32.0/35.6
Specific Impulse (sec)	269.4	250.8
Total Impulse (lbf-sec)	595	1289
Status	Developm.	Flight-Proven
Comments	Developed for GSFC for nanosats. High chamber pressure, low-eroding nozzle insert. Rel. long burn time	Long burn time motor for slow acceleration. Well suited for small spacecraft use.

A team of the Universities of Singapore and Arkansas took SRM design to an even greater extreme by building entirely MEMS fabricate SRMs out of silicon [30] – [32]. The planar thrust chamber and nozzle can be seen in Fig. 17. The silicon die is capped by a Pyrex 7740 cover through anodic bonding. An Au/Ti heater provides ignition. Propellant again is a composite of HTPB binder (14%), ammonium perchlorate oxidizer (74%) and added Al particles to increase performance [30]. Maximum performances obtained after several design iterations were 4.6 mN average thrust over a burn time of 169 ms, yielding a total impulse of 0.79 mNs [32].

It is difficult to foresee and application for such small motor as developed by Teasdale et al. [29] and Zhang et al [30] – [32] even for CubeSats. In fact, missions like Smart Dust, i.e. essentially free-floating chips in space may be the application for the types of micro SRMs.

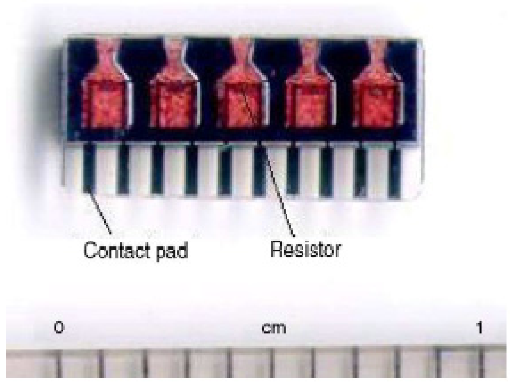


Fig. 17: MEMS Micromachined Silicon Solid Rocket Motors by Zhang et al. [31]

Digital Microthruster Arrays

Digital microthruster arrays extend the concept of micromachined solid rocket motors into arrays of potentially 100,000s of individual, single-pulse rocket motors arranged into an array across a micromachined silicon wafer. Such concept have been pioneered by teams at TRW (now NGST), Aerospace Corp. and Caltech [6], [33], [34], LAAS at CNES in France [6], [35] – [37], Princeton, Honeywell, Atlantic Research, and the University of Minnesota [6], [38], [39], KAIST in South Korea [40], and NASA Glenn [6], [41].

All concepts basically consist of three micro-machined wafers bonded into a stack, including a nozzle and burst disk, cavity, and igniter wafer. In the case of the LAAS group, the igniter is integrated with the burst disk. The fact that the thrusters are etched vertically into the wafer, as opposed to the planar design by Zhang et al [31] resulted in the typical square nozzle geometries through KOH etch with 35.3^{o} half angles. The NGST/Aerospace/Caltech design uses silicon nozzle/burst disk and igniter wafers and a FORTURAN glass wafer featuring cavities loaded with lead styphenate propellant, bonded using cyanoacrylate [34].

The LAAS/CNES design uses a silicon nozzle wafer, a second silicon wafer featuring the igniters and burst disks, and a third glass or silicon wafer holding the propellant cavities, loaded with glycide azide polymer mixed with ammonium perchlorate and doped with zirconium particles (GAP/AP/Zr) or zirconium perchlorate potassium (ZPP) [37]. An intermediate space wafer is bonded between the burst/burst disk wafer and the wafer holding the propellant cavities [37]. This was done to increase poor ignition rates by increasing the ullage volume per cavity, decreasing pressure rise time such that complete combustion could be achieved before the burst disk ruptures, increasing likelihood of ignition, completeness of combustion and thus array performance.

The Honeywell/Princeton concept uses a two stage silicon cavity wafer design, the first one featuring cavities filled with lead styphentae, triggering a combustion in a second wafer immediately bonded to the first, featuring cavities filled with a nitrocellulose mixture [39]. This mixture is burns very rapidly, allowing for complete combustion before the burst disk ruptures, increasing impulse bit performance (Fig. 18).

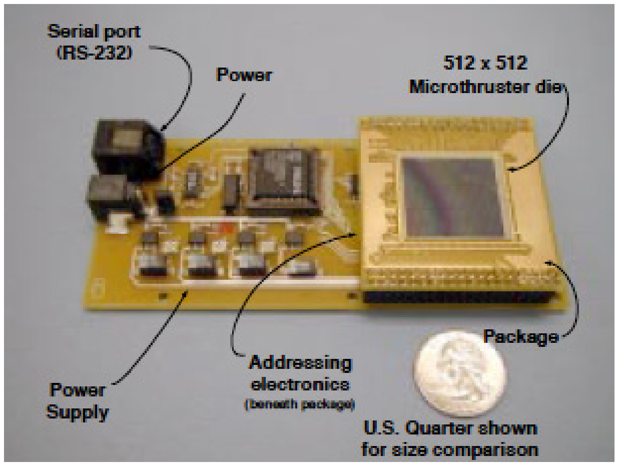


Fig. 18: Honeywell/Princeton Digital Micro-Thruster Array [39]

In the NASA Glenn concept, solid gas generator pellets were placed into cavities. The propellant was LAX 112 ($C_2H_4N_6O_2$) [6], [41]. Upon ignition, the product decomposes into nitrogen, hydrogen and oxygen. This concept uses no burst disk, reducing debris issues around the spacecraft.

Thruster performances for those devices where experimental data are available, are listed in Table 9. The NGST/Aerospace/Caltech design produced 0.1 mNs per cavity over a 1 ms pulse duration from arrays featuring 16 thruster cavities. Repeatability showed variations of up to 19% from cavity to cavity. The LAAS/CNES design produced up to 7.3 mNs using ZPP using a 250 μm nozzle throat over a duration of approx. 500 ms. Ignition energy for this propellant ranged between 17 to 24 mJ.

There is an obvious fascination in the microfabrication communities with digital thruster arrays given the large number of groups around the world working these thruster designs. However, upon closer inspection, this thruster design faces multiple implementation challenges. As indicated above, care has to be taken to ensure proper timing of burst disk rupture. Too early a rupture leads to incomplete combustion, poor performance, failure to ignite, or poor repeatability from cavity to cavity.

Once the disk does rupture, it creates a debris field around the spacecraft – the larger the more cavities are placed and fired per thruster wafer. While these generally will drift away from the spacecraft, they may pose a contamination risk in formation flying applications.

It has also been observed that igniting one cavity can set off multiple cavities through heat losses through the wafer structure, leading to uncontrolled firings of multiple cavities (Fig. 19). The designers have sought to address this issue through a variety of means. Using glass cavity wafers, instead of the highly thermally conductive silicon, reduces heat transfer. Youngner et al [39] placed silicon dioxide "firewalls" surrounding subsets of cavities to limit progression of uncontrolled firings across large sections of the wafer. Rossi et al [37] placed grooves surrounding each cavity to decrease thermal conduction. Such steps, however, decrease the packing density of cavities per wafer, limiting the number of firings available and the total impulse.

Even if operating properly, note that each thruster fires from a different position with respect to the spacecraft center line, each creating a different torque, placing major challenges to the attitude control system, in addition to similar challenges incurred by impulse repeatability. Delta-v applications would require at least two cavities to be fired opposite and at equal distances from the spacecraft centerline. Any repeatability uncertainties from cavity to cavity will induce torques, requiring additional attitude control. The total impulse per wafer, in particular when taking into account reduced packing densities to avoid uncontrolled firings, may also be low for delta-v applications.

Fig. 19: Unintended multiple cavity firing in a Digital Micro-Thruster Array die to heat conduction across the wafer [37]

Finally, the thruster wafers would have to be placed onto external wall surfaces of the spacecraft and thus occupy potentially valuable wall space needed for solar arrays, instruments, cameras, louvers, patch antennas and other equipment, especially at reduced cavity densities per wafer.

Digital micro-thrusters thus appear appear to pose several challenges to the system integrator. However, they may have an application niche for very small spacecraft, even smaller than CubeSats, where propulsion option are even more limited, and crude propulsive capability is needed for simple tasks not requiring much precision, such as initially dispersing a sensor net without the need for much control, for example, where no other propulsion options are available due to extreme size and mass constraints even exceeding those of CubeSats.

Bipropellant Thrusters

Bipropellant thrusters, as opposed to monopropellant thrusters, offer higher Isp performances approaching 300 sec for small, storable propellant engines, however, at the cost of added complexity and dry mass [6]. The propellant mass savings realized through the use of higher-Isp propellant combinations thus have to be balanced against the added dry mass of (at least) dual tanks for oxidizer and fuel, and of the mass the associated feed system branches and components (valves, filters, etc.). In order to maintain required mixture ratios throughout the mission, bipropellant engines are also pressure regulated, requiring a pressurization systems consisting of a gas supply tank and associated components (regulators, valves, filters, etc.). For this reason, a trade with monopropellant systems, which require only one (mono)propellant branch in the feed system and can be operated in blow-down modes, not requiring a pressurization system, outperform bipropellant engines in terms of system mass below delta-v's of approximately 500 – 1000 m/s depending on spacecraft mass. Cost of bipropellant systems are also always higher than for monopropellant systems due to higher engine complexity [6], part count of the feed system and integration. For microsatellites, given the higher dry mass fractions and more severe cost constraints, this trade will likely be pushed towards the higher delta-v end of this range, although it will always be a mission specific trade. At the higher delta-v end of this range it should be noted that much simpler and cheaper solid motors, available for microsatellite applications (see above) also become a competitor to bipropellant engines. The latter, however, do have an advantage when restartability is required.

Nonetheless, micro bipropellant rocket engine developments have been undertaken over the course of the last decade. Scharlemann et al [42], [43] of the Austrian Research Centre Seibersdorf have developed a conventionally, yet highly miniaturized engine and tested it on ethanol and gaseous oxygen [42]. Gaseous oxygen, while acceptable for initial experimentation, obviously cannot be used in practical applications as this would lead to the same system issues (high-pressurant storage, heavy, large tankage, leakage concerns) as for cold gas systems, although much higher specific impulse would obviously be achieved. Hydrogen peroxide was envisioned as oxidizer. The advantage of hydrogen peroxide is that it could also be used in a separate catalytic to power the engine turbines. Operation with hydrogen peroxide as oxidizer, however, was not achieved due to failure to ignite the engine, attributed to too short a combustion chamber length to facilitate sufficient mixing. The commercially procured turbopump used also was unable to provide sufficient chamber pressures [42]. It was concluded that a redesign of the engine is required [42].

Impressive gains in miniaturization have been achieved by London and Epstein et al. at MIT, who have developed an entirely MEMS-microfabrciated bipropellant rocket engine assembly machined entirely from silicon or silicon carbide [44 – 47]. The thruster chip consists of an integrated thrust chamber, nozzle, turbine pumps, and inlet valves. The entire assembly is about 20 mm square by 5 mm in height (Fig. 20). At this point, the entire engine assembly has not yet operated in this integrated assembly, rather each component was operated and tested by itself.

MEMS construction of this engine is complex. The thrust chamber features integrated cooling passage allowing for regenerative cooling of the thrust chamber [47]. The turbopumps, entirely machined out of silicon, run on fluid bearings. The inlet valves use electrostatically actuated pilot valves, which allow for fast actuation yet small stroke and actuation force [9]. These valves actuate main valves with 10 times the flow rate capability of the pilot valve. Ten of these valves are required to provide sufficient flow rate to the engine. Although operation on LOX and hydrocarbon propellants, as well as storable propellant combinations is being envisioned, at this stage thruster testing g was only performed with gaseous oxygen and ethanol. At a chamber pressure of 30 atm, thrust of 2.7 N (0.6 lbf) at equivalent 300 sec vacuum Isp values were obtained.

The high level of integration, once demonstrated at sufficient lifetimes and levels of reliability, alleviates some of the dry mass concerns outlined above. However, use gaseous oxygen will lead to similar concerns as for cold gas thrusters – the success of this engine thus rests on using liquid, and preferably storable propellants for in-space use on microspacecraft.

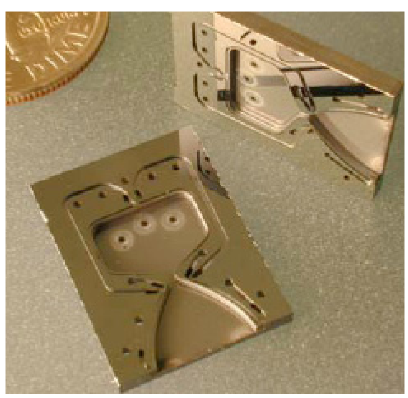


Fig. 20: MIT Micro-Bipropellant Engine Concept [47]

ELECTRIC PROPULSION OPTIONS

Pulsed Plasma Thrusters (PPTs)

Pulsed plasma thrusters were the first application of electric propulsion in space onboard the Russian Zond 2 spacecraft on its way to Mars in 1964/65 [48]. The spacecraft was ultimately lost due to a faulty radio. PPTs have been used since, in the West on the Navy's NOVA navigation satellites [6], [48], and finally the New Millennium Earth Observing (EO) – 1 spacecraft [6]. In a PPT, a capacitor, connected to two thruster electrodes, is charged and a discharge is triggered between these two electrodes. The ensuing arc ablates Teflon material off the solid fuel rod. This ablated material, due to Lorentz forces as a result of the current flowing through the plasma generated between the electrodes and its interaction with its surrounding. Self-generated magnetic field, accelerates and is expelled from the thruster [6], [48].

PPTs offer the capability of very small impulse bits, solid propellant storage in the form of Teflon fuel rods, modularity, and proven operation. As such, they inevitably became a technology of interest for microspacecraft applications [6], [49], [50]. Conventional technology, including the EO-1 thruster, are to massive for microspacecraft use, in particular Cubesats [6]. Mass drivers, accounting for 40 – 45% of the thruster system mass, are the power electronics and energy storage capacitors [49]. Focus on miniaturization of PPT technology thus has to focus on mass reduction in this area, and in particular on lighter capacitors designs. The thruster itself faces challenges during miniaturization, namely in the tailoring of discharge energy to the decreased fuel rod cross sectional area. If the discharge energy, as a result of overall system miniaturization, is too small, carbon neutrals in the plasma arc can return to the fuel rod surface an result in charring, which ultimately can lead to shorting of the thruster electrodes [51].

Recent advances in PPT miniaturization include the Aerojet “Dawgstar” thruster [49], [50], [52], [53] , the AF Micro-Pulsed Plasma Thruster (μPPT) by Spanjers et al. [51], a micro PPT developed by Austrian Research Centres Seibersdorf by Pottinger et al. [54], and a MEMS-based design by the Applied Physics Laboratory (APL) at Johns Hopkins University by Simon et al. [55] – [57]. The “Dawgstar” thruster is the largest of these, and was developed for an AF funded university nanosat project, utilizing spacecraft in the 15 kg class [52], [53]. As a result, its design is still recognizable as relative conventional, although significant strides where made in system components miniaturization, light-weighting, and system simplification. Given its intended target of somewhat larger microspacecraft, it may also be just too big for a CubeSat application. Its design, nonetheless reveals impressive gains in the miniaturization of this technology.

Shown in Fig. 21, the “Dawgstar” thruster system consist of four thruster modules with two thruster heads, both fed by a single capacitor per module. A single PPU charges the capacitors and triggers its discharges. Up to 4 capacitors can be charged simultaneously [52]. The “Dawgstar” thruster has delivered average 60 + 4 μNs impulse bits at thrust levels ranging from 61 – 119 μN, operating at 2 and 1 Hz. The thruster was operated as high as 3 Hz. It has accumulated 1.1 million pulses over 153 hrs of continuous run time at 2 Hz (Table 9). Additional tests brought the total number of pulses to 1.7 million pulses. The total system mass is 4.2 kg, incl. 4 thruster modules and PPU [52]. The single capacitor per module is a Mica capacitor rather than previously used oil filled capacitor, affording higher energy densities per mass [52]. Peak PPU inputs power ranges from 15.6 – 36 W depending on charging rates and thruster frequencies. The thruster efficiency is 1.8%, which is low, but low thruster efficiencies are typical for PPTs.

The AFRL μPPT is a novel concept, differing substantially from previous PPT designs, including the “Dawgstar” thruster, which still follows conventional designs. Rather than using spring-loaded fuel rods, propellant housing structure, separate electrodes and igniters, it uses a coaxial fuel rod [51]. Two types are being distinguished – the 2-electrode and the 3-electrode configuration. In the former, a conductive center rod is surrounded by an annulus of Teflon, contained in turn by an outer conductive rod, forming a coaxial thruster design. The center and outer conductive rods serve as electrodes, charged by a capacitor until surface breakdown self-triggers a discharge, followed by acceleration of the ablative material [51]. The Teflon surface recesses into the fuel rod, rather than being pushed forward by spring-loaded action as in conventional PPTs [6], [48], in turn consuming the inner conductive rod as well. The issue with this design is that self-triggered discharges is that large shot-to-shot variations exist. If discharge energies are too low, neutrals from the plasma can redeposit onto the Teflon surface, causing charring and eventually shortening of the two electrodes [51].

A three-electrode coaxial μPPT was thus devised. In this design, the outer conductive rod is surrounded by a second annulus of Teflon, in turn surrounded by yet another conductive rod containing the Teflon fuel. This design emulates the triggered conventional PPT designs featuring external igniters. In fact, the discharge between the inner and intermediate conductive rods are self-triggered as in the two

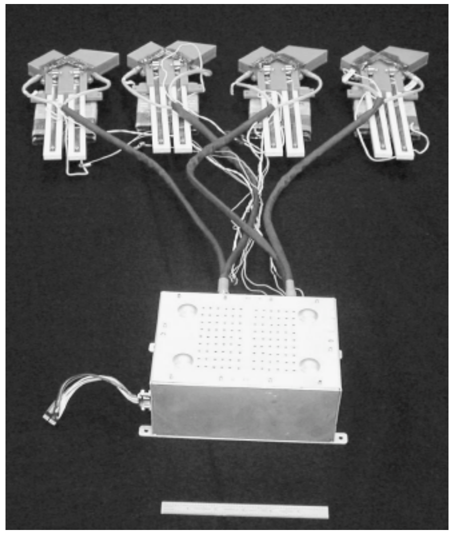


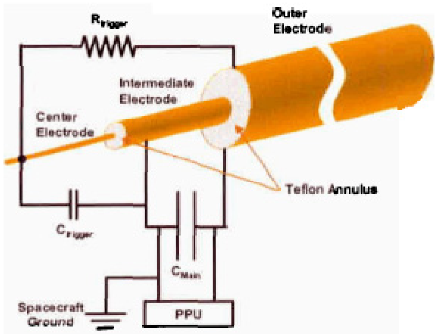
Fig. 21: The “Dawgstar” PPT Thruster System [52]

Table 9: Micro-PPT Performances

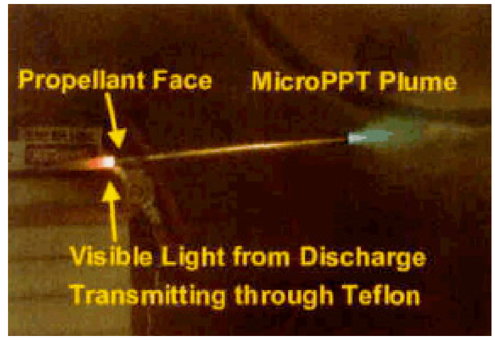
Parameter	Dawgstar	AFRL μPPT	ARCS μPPT	MILIPULT
Ibit (μNs)	60	2	2-25	0.4 – 0.6
Thrust (μN)	61-264	2-30		
Isp (sec)	266			
Mass (g)	4200 (system)	500 (system)	30 (5 g propellant)	13.5
Total Impulse (Ns)	1000			
Power (W)	15.6 - 36	1-20	0.5 – 4 (est.)	
Discharge Energy per Shot (J)	4.9	2.25		0.018 – 0.22
Capacitance (μF)	1.3 (per module)	0.42	2-6	0.937
Voltage (kV)	2.8	2.45 – 5.36		0.2 – 0.7
Frequency (Hz)	1-2			
Efficiency (%)	1.8			
Life (pulses)	1.7			
Propellant	Teflon	Teflon	Teflon	Water
Reference	[52]	[6], [51]	[54]	[55]

electrode design. The voltage between the intermediate and outer conductive rod is kept below surface breakdown voltages, and are instead triggered by the supply of seed ionization from the inner discharge, which now in effect acts as an igniter for the outer, or “main” discharge, which now operates in a more predictable “triggered” mode. As with the 2-electrode system, the inner electrodes are consumed along with the Teflon during firing, and the Teflon surface recedes into the coaxial housing. The AFRL μPPT can be seen operating in Fig. 22. The system weighs approx. 0.5 kg, delivered 2 μNs per shot at less than 1 J per pulse, and average thrust of 2 – 30 μN and 1 – 20 W of power [6], [48] (Table 9).

The Austrian micro PPT has demonstrated impulse bits as high as 90 μNs, with most data obtained in the 10-20 μNs range [54]. Careful examinations of fuel rod aspect ratios were conducted to optimize performances. Predicted performance ranges for this thruster are 0.5 – 4 W power, with a total impulse



(a)



(b)

Fig. 22: (a) AFRL μPPT Concept [51], (b) AFRL μPPT Firing in the Laboratory [51] – Plume to the right, discharge afterglow through the Teflon on the left.

capability of 49 μNs if a specific impulse of 1000 sec is assumed. Estimated mass for a flight like thruster is 30 g with 5 g propellant mass.

PPT miniaturization has been taken to an even greater extreme by Simon et al [55] – [57] using a planar, 2-D microfabrciated architecture. This design emulates the AF μPPT in that in uses a 2-D equivalent of the 3-electrode system, with the difference that electrodes are no longer conductive rods as in the AF design, but conductive strips of metal deposited onto a thruster chip. Three thin strips of copper are deposited onto G10 fiberglass or Alumina substrates, called the common, trigger and sustain electrodes. Triggering a discharge between the trigger and common electrode via a surface discharge following sufficient charging of the electrodes triggers a discharge between the sustaining and common electrode, which is held at voltage differences just below surface breakdown. As in the case of the 3-D AF coaxial thruster, this triggered electrode design allows for more predictable impulse bit performances, and eliminates the deposition of carbon particles from a low energy discharge, which could cause charring and shorting of the electrodes. Using Teflon tape deposited onto the substrate between the electrodes served as propellant.

However, frequent thruster failure was observed with these ablative Teflon thruster designs, likely due to leakage passages along the substrate material as a result of carbon deposition as a result of Teflon ablation [56]. The longest lasting thruster provided 10,000 pulses, most failed after shorter duration [56]. For this reason, the design was adapted for liquid propellant use. Termed the Micro Liquid Pulsed Plasma Thrusters (MILIPULT), it features a small liquid reservoir that allows flow to creep between the electrodes (Fig. 23). These experimental devices, however, featured to means to control the flow, especially between discharges [57]. For operational use this would be inacceptable, as it would mean that propellant would be lost through continuous leakage, reducing the effective Isp of the system and potentially posing contamination concerns to the spacecraft. Microvalves would be required to control the flow, posing significant challenges on their own. These valves would have to be chip based to match the envelope of this chip-based thrusters, be fast acting to allow for shot frequencies of one to several Hz typical for PPTs, and have lifetimes of potentially millions of cycles to match the PPT lifetimes, although these may be reduced for these smaller devices. As was shown in a previous survey by one of the authors [9], obtaining valve designs offering these features pose significant challenges. Electrostatic or piezovalves are the only allowing for such fast actuations and may still be offered in micromachined packages, but many details, such as leak tight seat designs, lifetimes etc remain to be worked out [9].

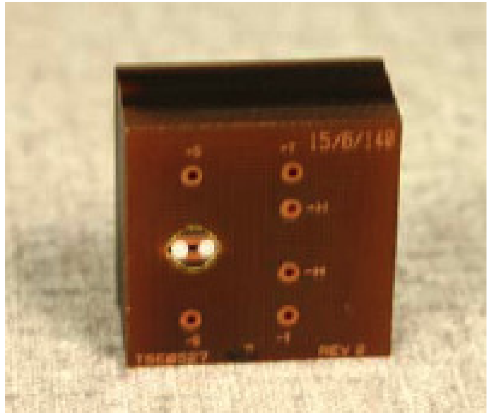
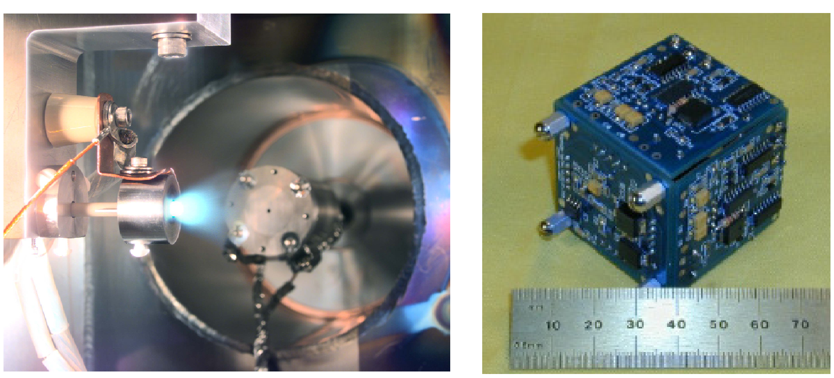


Fig. 23: APL MILIPULT Liquid Micro-PPT [55]

The MILIPULT thruster was tested without this feature, however, to for performance demonstration purposes. It was able to produce 0.4 – 0.6 μNs at 0.937 μF capacitances charge to 200 – 700 V using water as a propellant [55] (Table 9).

Vacuum Arc Thruster (VAT)

The Vacuum Arc Thruster (VAT) was developed by Almeda Applied Sciences Corp. (AASC), and is another ablative pulse plasma type, yet using metal electrodes [58] – [61] (Fig. 24). Applying a high voltage between tow electrodes leads to electric field intensification near surface roughness spikes of a metal electrode. The high power density in this spike leads to vaporization and ultimately electric breakdown and plasma generation in an arc discharge. Different cathode materials can be used, including Carbon (C), Aluminum (Al), Tungsten (W), Bismuth (Bi), or Chromium (Cr), among others. The plasma plume is then allowed to expand into vacuum and produces a thrust pulse. Impulse bits vary with electrode material chosen, and can range from 1 μNs for Cr [59], 30 μNs for W [58] to 40 μNs for Bi [58]. Larger impulse bits can be built up by rapid firing of discrete impulse bits, ablating multiple roughness spikes. Up to 200 Hz firing sequences have been demonstrated. Test on a JPL micro-Newton thrust stand demonstrated up to 100 μNs for 100 Hz pulse trains using Cr [59].



(a) (b)

Fig. 24 (a): VAT Thruster firing on the JPL Micro-Newton Thrust Stand, (b) VAT PPU configured for CubeSat [59]

A CubeSat VAT module was developed, making the VAT, together with the PPY and the chemical VACCO MiPS system, one of the few (near) off-the-shelf thruster systems available for CubeSat propulsion today. In a past design study, which delivered thruster hardware intended for use on the ION-F CubeSat, a 150 g PPU was built, able to supply 4 VAT thruster heads with power using a 12-24 V bus voltage [59]. Thruster heads are estimated to weigh 100 grams and average power, depending on thrust values needed, may range between 1 – 100 W [59]. Thrust to power values of up to 10 μN/W are expected to be obtained from such a thruster.

The advantages of the VAT are its very low impulse bits for microspacecraft attitude control, solid propellant storage, with the propellant being integrated into the thruster head, as for conventional Teflon PPTs, thus allowing for inherently compact and modular designs. Given the small impulse bits per ablative pulse, however, even high cycling frequencies limit total thrust values that can be obtained, and large delta-v applications may not be possible, despite high specific impulse that can be achieved with this thruster, ranging from 1000 – 3000 sec [58],[59] depending on propellant. Rapid maneuvering, as in the case of an inspector sat, may also be limited for the VAT due to its low accumulative thrust values. Chemical systems, such as the VACCO MiPS may be more suitable for that application. The VAT may, however, be ideally suited for precision pointing and attitude control.

Miniature Ion Engines

In an ion engine, a plasma is generated in a discharge chamber, and ions are extracted from it via an electrostatic grid system. Typically xenon is being used as propellant gas, although historically mercury vapor has been used, as well as other inert gases, such as argon and krypton. Two main types of ion engines are being distinguished, depending on how the plasma is generated [6], [62]. In a DC electron bombardment, or Kaufmann-type thrusters, a cathode (typically a hollow cathode) is used to generate a plasma from which electrons are extracted and injected into the discharge chamber to ionize the propellant gas by accelerating them towards an anode surface. Permanent magnets mounted near the anode reduce electron loss and increase the electron path through the discharge, and increase engine efficiency. The hollow cathode is fed the same propellant as the main discharge. In an radio-frequency (RF) ion engine, no cathode is required for plasma generation, rather, a magnetic coil wrapped around a dielectric discharge chamber generates an oscillating, predominantly axial magnetic field that according to Maxwell's laws generates an oscillating azimuthally electric field following the curvature of the cylindrical discharge vessel. Electrons present to small degrees within any gas are accelerated, cause ionization, and additional electrons are provided in these reactions to eventually cause plasma generation. Electron energy cost per thrust-producing ion generated by an RF thruster are typically higher than for the DC discharge, but no life limiting engine cathode is required other than the neutralizer. This neutralizer, in both cases (RF and DC), consists of another hollow-cathode mounted externally to the engine, providing electrons for beam neutralization of the positively charged ion beam, thus avoiding spacecraft charging.

Several miniature ion engines have recently been developed, although not specifically for CubeSat applications, but rather for formation flying applications on future space telescope [63], [64]. The use of inert xenon propellant near sensitive optical surfaces, as well as the ability to smoothly modulate the thrust amplitude rather than using a chemically or PPT based pulsed thruster approach, or reaction wheels, which each may induce jitter, have made miniature ion engine a critical component of future development for theses types of missions.

As for conventional ion engines, the two types (DC and RF) have been miniaturized for these application. Micro-ion engines were recently re-proposed by Janson at Aerospace Corp. [6], [65], and by Brophy et al at JPL in the form of an ion thruster on a chip (ITOC) concept [66], although miniature RF engines existed decades before that (see below). Initial studies at JPL of this ITOC concept included critical subcomponent evaluations to study the feasibility of this approach, including microfabrciated accelerator grid testing [67] – [69] and field emission based cathode testing for discharge generation and neutralization [70], [71]. The ITOC thruster concept eventually morphed into a 3-cm diameter meso-scale version, appearing more feasible and useful to future applications of this technology [72] – 78]. Termed the Miniature Xenon Ion Thruster (MiXI) its primary goal for application became to the Terrestrial Planet Finder (TPF), and similar exoplanet missions [63], [64], [76]- [78]. A laboratory prototype of MiXI is shown in Fig. 25 (a) and (b). Table 10 summarizes recently demonstrated performances of this engine, compared to other devices reviewed in the following. MiXI has demonstrated up to 1.5 mN thrust at 50 W power and a specific impulse of 3200 sec on xenon propellant. Thruster mass is 200 g.

RF-ion engines are attractive candidates for miniaturization as no internal cathode is required. A 4-cm diameter RF ion thruster was originally developed at Giessen University in 1972, far ahead of its time [79], but this concept was abandoned for several decades. As in the US, recent interest in formation flying missions in Europe led to a reactivation of the 4 cm engine concept, as well as smaller RF ion engines since the middle of the present decade [79], [80]. Figure 26 shows a 2.5 cm dia RF ion engine named the μNRIT-2.5. Its performances are summarized in Table 10. The 2.5-cm device weighs 210 gram, comparable to MiXI, and delivers 0.5 mN of thrust using 30 W of power using xenon.

RF-micro-ion engines have also under development in the US in recent years. A Miniature Radio Frequency Ion Thruster (MRIT) of 1 cm diameter has been under development at The Pennsylvania State University (Figure 27) [81]. Initial laboratory testing was conducted on argon, however, future operation is envisioned using xenon. Performances demonstrated with MRIT are listed in Table 10 along with MiXI and the 2.5-cm and 4-cm μNRITs. MRIT has demonstrated 59 μN of thrust and 5480 sec Isp at an RF power level of 15 W.

(a)

(b)

Fig. 25 (a) JPL Miniature Xenon Ion Thruster (MiXI), (b) MiXI in Operation

Table 10: Micro-Ion Engine Performances

	MiXI	μNRIT-2.5	MRIT
Thrust (mN)	0.01 – 1.5	0.05 -0.6	0.001 – 0.06
Isp (sec)	2500 - 3200	2861	5480
Power (W)	13-50	13-34	
Electrical Efficiency (%)	> 40	4 - 47	15
Mass Utilization (%)	> 70	15 - 52	
Dia (cm)	3	2.5	2
Mass (g)	200	210	
Propellant	Xenon	Xenon	Argon

Fig. 26: The μNRIT-2.5 Miniature Ion Thruster of the University of Giessen [79]

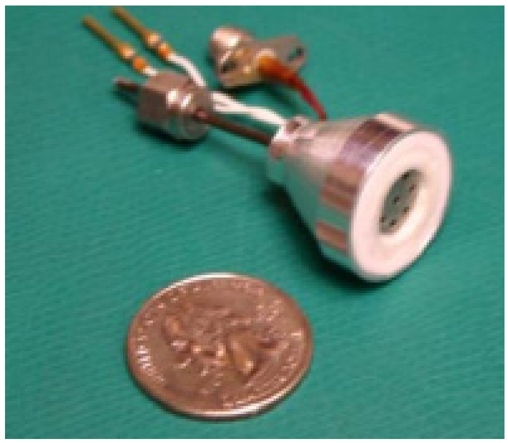


Fig. 27: The MRIT Micro-Ion Thruster of Penn State University [81]

Miniature ion engine technology, although developed for formation flying missions, could potentially be adapted to CubeSats or similar microspacecraft and deliver large delta-v capability to microspacecraft, providing significant orbit raising, multiple orbital plane change, and extended drag-make up capabilities in Earth orbit, thus allowing for significant and extended maneuverability and associated mission flexibility, and even allow CubeSats to eventually venture beyond Earth orbit. In the case of the latter it must be noted, however, that stand-alone microspacecraft use beyond Earth orbit, due to communication link issues from very small spacecraft platforms like CubeSats to Earth, may be limited. A more likely scenario for microspacecraft use beyond Earth orbit may be their deployment from larger spacecraft for certain aspects of an interplanetary mission, e.g. for particularly risky parts of a mission, such as the in-situ exploration of Saturn's rings, for example, or widely distributed sensor nets requiring out of ecliptic maneuvers [6]. Even in these cases, delta-v requirements may still be significant, and may be benefitted the development of high-Isp thruster options such as micro ion engines.

To realize micro-ion engine propulsion onboard CubeSats, in addition to reducing the size of the engines, other required subsystems have to be equally miniaturized, including feed systems and power processing unit (PPU). VACCO's ChEMS module approach has already been adapted to xenon flow controllers, albeit for conventionally sized engines, but could be adapted for micro-ion engines on CubeSats given its already demonstrated feasibility of integration on CubeSats under MEPSI (above). If miniaturized light weight PPUs could be developed as well, a "3U" CubeSat architecture may be envisioned, consisting of one "1U" module featuring the micro-ion engine an a micro-feed system, the second "1U" unit hosting the PPU, and the third "1U" unit holding the payload and other spacecraft subsystems. The PPU unit would also require deployable arrays to provide power levels in the range of several 10s of Watt as required for micro-ion engine systems.

Miniature Hall Thrusters

In a Hall thruster electrons emitted by a cathode external to the thruster are accelerated towards an anode at the end of an annular channel. On their way, they cross a radial magnetic filed near the entrance of the channel. Due to Lorentz force action, they gyrate around the magnetic field lines and drift azimuthally through the annular channel, in the course of which they cause ionization. The ions are accelerated out of the thruster by the same field that accelerated the electrons. Due to the higher inertia of the ions, they are significantly less effected by the magnetic field, and can exit the channel to produce thrust. Electrons from the external cathode neutralize the beam. The high electron density in the annular channel reduces space charge effects as in an ion engine, thus a more compact engine design can be realized for a given ion beam [6].

Hall thrusters typically deliver a specific impulse in the 1500 – 2000 sec range [6], [62]. This makes them generally more suitable for near-Earth missions. Optimization of electric propulsion devices requires balancing the propellant mass savings realized by the higher specific impulse of the device with the power requirements and associated masses (for power generation and conditioning) to generate this higher specific impulse. For the more moderate delta-v requirements in Earth orbits, as compared to interplanetary missions, additional propellant mass savings by raising the specific impulse significantly above a level of approx. 2000 sec are no longer offset by the higher power system masses. Hence, a Hall thruster system operates more efficiently in Earth orbit than an ion engine. Thus optimization strictly only holds for state-of-the-art conventional electric propulsion technologies, as deliverable specific impulses, efficiencies, and specific power values used in this optimization are tied to existing technologies. Micro technologies may have different performance characteristics, which may shift the optimum. Given that performance parameters of micro propulsion and power technologies, still being under development, are just emerging, however, it seems reasonable to consider micro Hall thrusters as a potentially interesting electric propulsion option for Earth-orbit operations of microspacecraft as well.

Hall thrusters, however, face particular challenges in their miniaturization given their plasma generation scheme described above [6]. Reducing the size of the thruster requires to reduce the Larmor radius (i.e. radius of curvature of the electron trajectories in the magnetic field) to avoid excessive wall impingement, and hence electron wall and efficiency losses. This in turn required increased magnetic field strengths that need to be realized as the overall thruster size is reduced. Furthermore, increased heating of the magnets by the plasma is to be expected in smaller devices, which could lead to demagnetization, and loss of magnetic field strength, and hence electron containment.

Consequently, initial attempts at Hall thruster miniaturization were fraught with significant challenges. A miniature Hall thruster concept by MIT [82], 4 mm in dia., delivered 1.8 mN thrust and 826 sec specific impulse at 126 W of power as shown in Table 11 [6], [82]. Efficiencies were poor at only 6% due to the aforementioned demagnetization issues (Figure 28).

Fig. 28: MIT 50-W Hall Thruster [82]

Table 11: Small and Micro-Hall Thruster Performances

	BHT-200	SPT-30	MIT	PPPL CHT 2.6	PPPL CHT 3.0
Thrust (mN)	4-17	5.6 - 13	1.8	2.5-12	3 - 6
Isp (sec)	1200 - 1600	576 - 1370	865		1100 - 1650
Power (W)	100 - 300	99 - 258	126	50-300	90 - 185
Efficiency (%)	20 - 45	16 - 34	6	15 - 32	20 - 27
Mass (g)	<1	`~1			
Dia (cm)	2.1	3	0.4	2.6	3
Propellant	Xenon	Xenon	Xenon	Xenon	Xenon
Reference	[87]	[6]	[82]	[83]	[84]

A novel type of miniature Hall thruster developed at Princeton Plasma Physics Laboratory (PPPL) overcomes several of the issues in Hall thruster miniaturization, and specifically the low efficiencies observed in previous versions. The PPPL type uses a cylindrical Hall thruster (CHT) architecture, which distinguishes itself from the conventional annular Hall geometry by an open cylindrical, rather than an annual flow channel and a largely axial, rather than radial magnetic field [83] – [87]. Electrons are once again emitted by an external cathode, and accelerated towards an anode located at the innermost end of cylindrical channel. However, the axial magnetic field is concentrated at the anode end of the channel and therefore features a strong axial gradient near the anode, causing electrons to be mirrored back away from the anode, as they approach it. In a CHT, a region of high electron density is thus being generated due to the balance of attracting electrostatic forces due to the anode, and the mirror effect of the concentrated magnetic field (Figure 29) [84].

Cylindrical Hall thrusters of 9-, 3-, and 2.6-cm diameter have been built and tested. The 9-cm devices is not a miniature Hall thruster type, but sized for a power level on the order of 1 kW. Performances of the 3- and 2.6 cm devices are listed in Table 11. Significant is the increase in efficiency ranging into the 30%. The 2.6-cm thruster delivers up to 12 mN at 300 W, whereas the 3 cm thruster, operated at lower power levels, delivers 3 – 6 mN at power levels from 90 – 185 W and specific impulses from 1,100 - 1,650 sec specific impulse on xenon. Also shown in Table 11 are BHT-200 Hall thruster performance data by Busek for comparison. The BHT-200 is currently the state-of-the-art of existing, commercially available Hall thruster technology, and was the first US Hall thruster operated in space onboard Tac-Sat 2 in Dec. 2006.

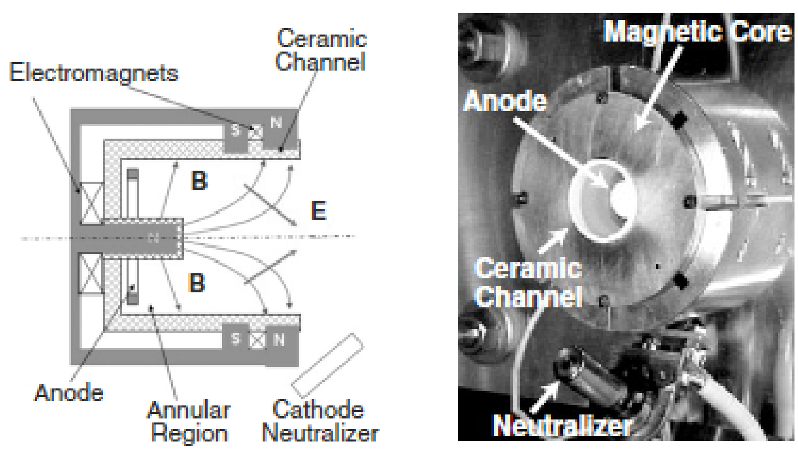


Fig. 29: Cylindrical Hall Thruster (CHT) by PPPL, (a) Concept, (b) 3-cm CHT device [84]

Integration of miniature Hall thrusters systems into CubeSats faces similar challenges as for ion engines, i.e. miniaturized feed systems, power conditioning units and deployable solar arrays to provide the necessary power. Power processing unit miniaturization may be aided by the fact that typically only 300 V anode voltage are required, vs in excess of 1 kV for ion engines, and fewer power supplies are needed for Hall thrusters than for ion engines. If these can be realized, a 3U CubeSat bus may accommodate such a system and provide significant delta-v capability for near-Earth missions.

Electrospray Thrusters – Colloid

As becomes evident when surveying ion and Hall thruster miniaturization, significant challenges arise from the miniaturization of the plasma chamber, and efficiency losses associated with it. Electrostatic thruster devices do exist that do not require a plasma discharge, and emit ions or charged directly droplets directly from a propellant columns based if a strong electrostatic field is applied [6]. In an electrospray thruster, propellant is fed up an emitter, which can take the form of a sharp needle, capillary, or narrow slit. An electric field is established between the emitter and an opposing electrode. The field is intensified near the emitter tip and, depending on the electrically conductive propellant is distorted into a sharp so called Taylor cone as a result of a balance of surface tension and electrostatic forces. The shape of this Taylor cones intensifies electric field strengths even further. Depending on the type of propellant, either ions are extracted from the propellant directly by field emission, as in the preferred case for liquid metal propellants, or breaks up into charged liquid droplets in the case of doped glycerol propellants. Liquid metal ion sources (LMIS) are typically also referred to as Field Emissions Electric Propulsion (FEEP) and are reviewed in the next section. Thrusters relaying primarily on droplet emissions are commonly referred to as colloid thrusters and are the subject of this section. Note that the difference between these two thruster types is less well defined in the case of newer, so called ionic liquids, which are essentially molten salts at room temperature, and can both form ion and droplet emission depending on operating conditions, such as flow rate, temperature and applied voltage [88]. Not requiring the generation of a plasma discharge, these thrusters are extremely compact, and given the sub-millimeter sizes of emitter tips, lend themselves well to miniaturization [6].

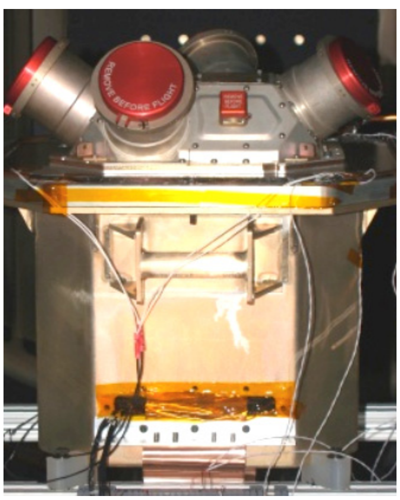
Colloid thrusters were extensively studied in the US in the 1960s by Perel and Mahoney [6], [89] – [91], and at TRW (now NGST) [92], [93]. Research focused on glycerol propellants, doped with sodium iodine or sulfuric acid – the latter producing negative charged droplets allowing for a bipolar thruster operation without the need for a neutralizer. Demonstrated thruster performances ranged from 1 – 334 μN per emitter at specific impulses from 450 – 1,450 sec at specific power levels ranging from 4 – 10 W/mN [6]. The highest specific impulses were achieved at very large emitter voltages approaching 20 kV [6]. Emitter sizes ranged from 100 μ to over 2 mm for the higher thrust devices [6]. Data obtained with these historic thruster devices are summarized in detail in Ref. [6].

Interested in colloid thrusters waned over the ensuing decades as spacecraft grew larger, and no application fro colloid thrusters materialized. This changed dramatically in the late 1990s and early 2000s, as two spacecraft applications emerged in need of either precision thrust or micropropulsion devices, namely formation flying missions and microspacecraft. Initial developments were primarily focused on microspacecracft applications. Perel et al [94] re-engaged into colloid thruster development with a 1 μN I.D. capillary emitter and Pranajaya et al [95] developed a 50 μN I.D. capillary device with up to 100 emitters for a university nanosat mission. The latter produced up to 1 μN per emitter at an estimated specific impulse of 500 sec and a specific power value of 10 W/mN [95]. The overall thruster module, including feed system and PPU under development has a mass target of 0.5 kg at a size of 10 x 10 x 20 cm^3 [95]. Ac such it would fit a 3U CubeSat bus and leave 1U for payload and other spacecraft subsystems.

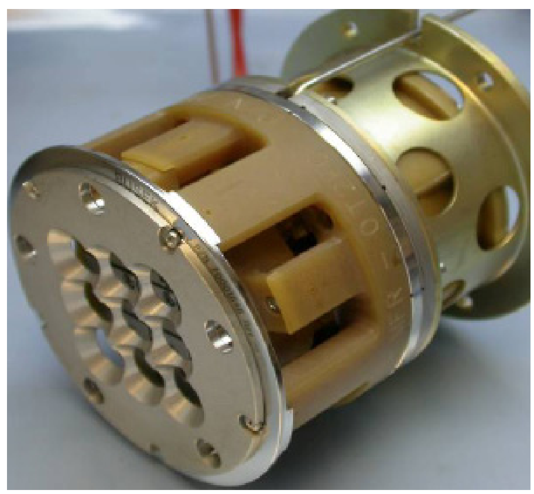
Colloid thruster developments made significant further strides shortly after these initial renewed forays into colloid thruster development through NASA's New Millennium ST-7 and Laser Interferometry Space Antenna (LISA) flight programs. NASA selected a colloid thruster system developed by Busek, in collaboration with JPL for these missions [96] – [98]. LISA requires three spacecraft to fly in a triangular formation, spaced approximately 100,000 km apart, allowing gravity waves, should they exist, to be detected by moving the spacecraft with respect to each other, and detecting these movements by laser interferometry. In order to distinguish between potential gravity wave effects and other disturbances forces, such as solar wind, for example, the spacecraft are required to precisely follow a spacecraft internal proof-mass to within 10 nm. Micro-Newton levels of thrust to a precision of 0.1 μN are required to accomplish this task.

The US is contributing a colloid propulsion system to accomplish this task. The thruster system consists of multiple clusters featuring for thruster heads clocked 90 degrees with respect to each other provide precision control (Fig.30). The cluster module also houses the propellant supply and feed system, consisting of compressible bellows housing the propellant to pressure-feed each thruster head, and the PPU. Each thruster head has demonstrated a thrust range of 5 – 38.5 μN of thrust to a precision of less than 0.01 μN [96]. Specific impulses ranges from 190 – 240 sec depending on propellant temperature. Temperature control is achieved with individual thruster heaters, driving the power per cluster to approx. 24 W during warm up, and 16 W during normal operation. The ST-7 colloid thruster performance characteristics are listed in Table 12.

Obviously, the cluster is too large for CubeSat applications, however, each thruster head could form a stand-alone thruster for a CubeSat for delta-v applications. It currently features 9 individually machined emitters (Fig. 31), the number of which could be tailored to the desired application. A dedicated feed system and PPU would be required, and miniaturization challenges of a colloid system are likely to focus more on these two subsystems than the thruster head itself, an experience already made by Pranajaya et al. [95]. Using ionic liquids, operating colloid thrusters in both droplet as well as ionic modes would allow



(a)



(b)

Fig. 30: (a) ST-7 Colloid Thruster Cluster [97]; (b) ST-7 Cluster head (4 per cluster) [98]

Table 12: ST-7 Colloid Thruster Performance Characteristics [96]

Parameter	Demonstrated Performance
Thrust (μN)	5 – 35.8
Thrust Noise (μN)	<0.01
Isp (sec)	240
Mass (kg)	15
Power (W) - Max	24.6 (all thruster heaters at max at 2 W each)
Power (W) - nominal	16
Beam Divergence (Half Angle) (degree)	<23

Fig. 31: Individual ST-7 Colloid Thruster Emitter [98]

such a system to cover a wide range of specific impulse from a couple hundred to a few thousand second specific impulse, allowing for substantial maneuverability of a CubeSat equipped with it, from high thrust-to-power low Isp modes for rapid maneuvering, to low thrust-to-power in a high Isp mode for large delta-v maneuvers.

Electrospray Thrusters – FEEP

FEEP thrusters function similarly to colloid thrusters, as described above, however, emit predominantly ions. External wetted needle or internally wetted slit emitters are commonly used [6]. The former use indium, the latter cesium as a propellant. Unlike colloid thrusters, FEEP thrusters are not pressure fed, and propellant is supplied to the emitter tip by capillary forces, making for very compact feed systems featuring no valves or pressurized propellant tanks.

Field emission electric propulsion was initially studied in the US by Perel et al. in the 1960s [90], [91], focusing on capillary tubes to feed cesium propellant. As with colloid thrusters, research into FEEP thrusters in the US waned as spacecraft grew larger, and no unique applications requiring FEEPs materialized. Unlike colloid thrusters, however, FEEP development was continued in Europe, initially at ESA-ESTEC in The Netherlands, then the University of Pisa and Centrospazio, later Alta in Italy as well under funding from ESA [6]. This work focused on cesium FEEPs and ultimately led to the development of slit emitter designs of varying slit widths, ranging from 2 mm to 70 mm producing thrust from 40 μN to 1.4 mN. Two relatively recent FEEP slit emitter designs are listed in Table 13. Specific power values are 66 W/mN – considerably higher than for colloid thrusters due to high Isp operation in the cases of these thrusters around 9,000 sec. The FEEP-5 thruster is shown in Fig. 32.

Fig. 32: FEEP-5 Thruster by Centrospazio/Alta

Table 13: FEEP Performance Characteristics

Supplier	Designation	Thrust (mN)	Mass (kg)	PPU Mass (kg)	Isp (sec)	Power (W)
ARCS	In-FEEP 100	0.001 – 0.1	0.3	Incl.	8000 - 12000	0.5 - 10
ARCS	GOCE MTA	0.002 – 0.65	3.5	Incl.	8000 - 12000	6-52
ARCS	In-FEEP 1000	0.001 - 1	1.5	Incl.	8000 - 12000	2-80
Centrospazio	FEEP-5	0.04	0.6	1	9000	2.7
Centrospazio	FEEP-50	1.4	1.2	1.2	9000	93

Cesium FEEP thrusters suffer from complications in the propellant handling, as cesium is very corrosive and readily reacts with even minute amounts water, oxygen, or CO_2, forming cesium hydroxides, oxides, and carbonates with high melting temperatures, which may thus clog emitter slits [6]. A lid mounted in front of emitter slits need to isolate the thruster from exposure to the environment, and may only be opened in space. Subsequent to the opening of the lid the thruster must immediately be fired again to avoid contamination of the cesium at the emitter exit slit due to spacecraft outgassing.

A significantly more benign propellant is indium. Solid at room temperature up to its melting point of 154 C, it requires heater power to liquefy it and induce thruster operation, however, can be handled under atmospheric conditions, thrust greatly simplifying propellant handling [15]. Its very low vapor pressure and lack of corrosiveness as exhibited by cesium also reduces spacecraft contamination concerns, which is of particular concern for microspacecraft, if they are to be used information flying missions with the potential of cross contamination due to neighboring spacecraft intersecting thruster plumes.

The Indium FEEP or LMIS (Liquid Metal Ion Source) was developed at Austrian Research Centres Seibersdorf over 20 years ago, although originally not as a propulsion devices but an ion source [15]. In this function, it has seen flight applications onboard the Russian MIR space station in 1991 as part of a mass spectrometer experiment, and subsequently on multiple other space missions for spacecraft potential controls and in mass spectrometers [15].

Since then development has focused on developing Indium LMIS for space propulsion applications. Several devices recently under development at ARCS are listed in Table 13. Single emitter thrust performances typically range up to 15 μN – higher thrust levels can be achieved by assembling arrays of multiple emitters [15]. A 100-μN device was originally developed for the ST-7 mission, and featured 9 emitters (Fig. 33) [15]. An array for the European GOCE mission required 16 emitters (Fig.34).

CubeSat integration of In FEEP would require accommodation of the PPU, and additional deployable solar arrays as other electrostatic thruster devices reviewed above. Feed system integration, however, is extremely simplified, as the propellant reservoir and capillary feed are part of the thruster head. FEEP thruster allow for extremely high specific impulse values, however, at reduced thrust-to-power ratios, requiring either long transfer times or high power levels. At present, droplet emission with Indium thrusters has not yet been demonstrated in a controlled fashion. Were it feasible, a dual mode operation as for colloids operating on ionic liquids (above) could greatly extend mission flexibility.

Fig. 33: ARCS In-FEEP-100

Fig. 34: ARCS GOCE MDA Array [15]

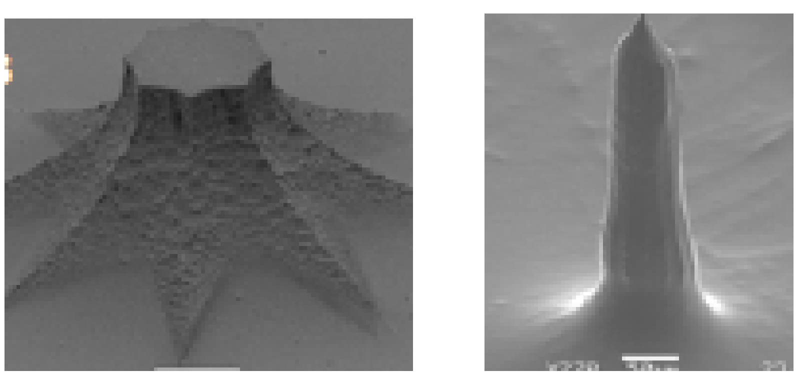
Microfabricated Electrospray Arrays

Inspecting Fig. 34, it becomes obvious the integrating emitters by conventional means is not a promising solution to arrive at large arrays. Large arrays, however, may be desirable to scale the obtainable thrust from electrospray devices for a variety of applications – extending beyond the current focus on precision pointing of formation flying spacecraft to delta-v applications as well, thus fulfilling multiple functions with a single type of thruster, simplifying propulsion subsystem architecture. The large conventional array sizes are in particular disappointing as the critical thruster features – i.e. the emitter tips – are already sub-millimeter, even micro—sized in many applications. While cesium slit emitter thrusters can easily be scaled up by lengthening the slit, and in fact thrusters capable of milli-Newton thrust levels have been built [6], use of cesium is not really desirable for reasons stated above.

A more efficient way to integrate emitters into arrays is through the use of microfabrication techniques [6]. In fact, the first ever proposed MEMS thruster concept was conceived by Mitterauer in 1991 in the form of a microfabricated FEEP concept [6], [99]. No devices were ever built, however. The idea resurfaced again in 1998 through a proposal by Marcuccio et al [100]. Initial forays into the experimental study of microfabrciated colloid and FEEP thruster concepts were made at JPL under small discretionary funds with initial focus on the wetting abilities of silicon substrates [75]. Work on microfabricated electrospray arrays gained momentum with significant microfabrication and testing efforts at MIT in collaboration with Yale [101] – [106], the University of London in the UK [107] –[113], after initial efforts at the University of Southampton there [107], and Sandia, in collaboration with ARCS of Austria [114], [115]. Initial work was also conducted at the NASA Goddard Space Flight Center (GSFC), but not continued [116]. Quite early this concept gained attention in China, were a couple of unique microfabrciated electrospray arrays were developed and tested in the early part of the last decade [117], [118]. So far unpublished work was subsequently sponsored by DARPA at SEI Inc., and at Almeda Applied Sciences Corp (AASC) in a joint effort with JPL and the University of California at Irvine (UCI), and finally presently at JPL.

The MIT/Yale work made significant contributions both in the fabrication and testing of silicon micromachined electrospray arrays [101] – [106], as well as the development of novel electrospray propellants [88], namely ionic liquids, which demonstrated dual mode operation (ionic and droplet). Initial development efforts focused on linear arrays [102], [105], featuring a self-aligning cathode attaching to the emitter array via a unique silicon micromachined “clip” mechanism (Fig. 35). Limited packaging densities achievable with linear arrays led the team to explore 2-D arrays architectures [105], [106]. A unique contribution was the development of “black silicon”: employing a chlorine plasma etch, the surface of a silicon substrate is roughened, aiding wettability. It is called “black” because of its appearance as the result of the etch and ensuing roughening [105]. The MIT group also explored grooved emitter tips for the first time, called “truncated volcanos” (Fig. 36 (a)), as well as more traditional “pencil” shaped emitters (Fig. 36 (b)) [106]. The emitter grooves, however, were not designed to aide emitter wetting, but to produce sharp tips at the upper surface of the truncated emitter tips to serve as emission sites [106]. Volcano emitters were 270 μm tall and had an upper truncated surface diamater of 90 μm. The radius of curvature of the tips was 6 μm [106]. Pencil shaped emitters were also 270 μm tall, had a diameter of 50 μm and a tip radius of curvature of 3 μm [106]. Emitters were spaced 200 μm apart. Pencil emitter arrays of up to 1025 emitters have been fabricated and were shown qualitatively to emit using EMI-BF_4 ionic liquid, as evidenced by markings left on an opposing electrode [106]. Volcano emitter arrays, featuring 2 x 2 as well as 8 x 8 8-tip truncated volcano emitters have also been tested and emitted currents in the range of 2 x 10^{-4} to 2 x 10^{-1} μA of current per emitter using the same propellant.

Fig. 35: Linear Microfabricated Colloid Thruster Array by MIT [102]



(a) (b)

Fig. 36 (a) Micromachined “Truncated Volcano” Emitters and (b) Pencil Shaped Emitters by MIT [106]

In the British development, initially capillary emitter arrays were used [108]. Up to 20,000 emitters have been fabricated per 100-mm dia. silicon wafer (Fig. 37) [108]. Initial experiments using the traditional glycol propellant doped with sodium iodine, used 400 μm O.D. and 100 μm I.D capillary emitter tips and produced between 300 – 450 nA per emitter at voltages between 7.3 kV and 8.6 kV and various flow rates ranging from 54 nl/s to 151 nl/s [108]. More recently, emission using EMI-BF_4 propellant was demonstrated first using a single 30 μm silica tip, then with silicon micromachined 19-emitter arrays, and dual mode operation was observed, likely due to ionic and droplet emissions [111]. With the aide of the Ecole Polytechnique Federale de Lausanne (EFPL) in Switzerland, unique bead filled silicon micromachined arrays were fabricated (Fig. 38) [113]. By filling the capillaries with beads, hydraulic resistance can be tailored to provide the appropriate flow rates. Once again, 19 emitter arrays were built with capillaries 70 μm in height, 34 μm O.D and 24 μm I.D. Silica beads of 4.74 μm average diameter were filled into the capillaries via an alcohol based carrier fluid, which was subsequently evaporated. Heating the assembly fused the silica beads via a silanization process [113]. Both EMI-BF_4 as well as EMI-Im ionic liquids were tested. Using these array with EMI-BF_4, specific impulse values as high as 3500 sec was measured at an emitter voltage of 1.2 kV and at thrust levels of 40 pN, averaging to about 2 pN per emitter [113].

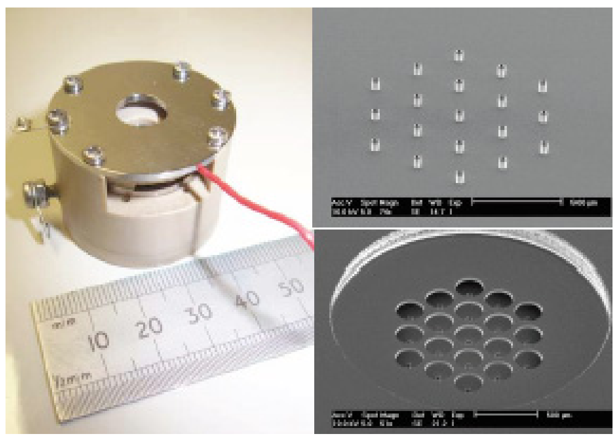


Fig. 37: Micromachined Colloid Emitter Array by the University of London [111].

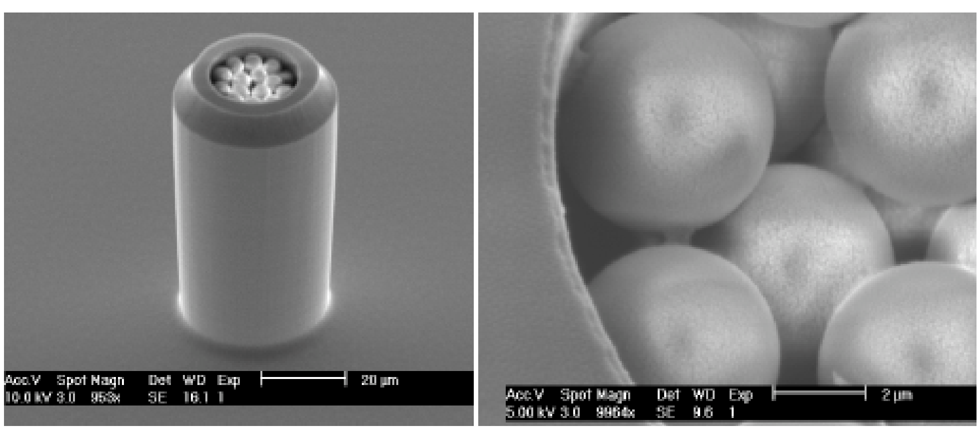
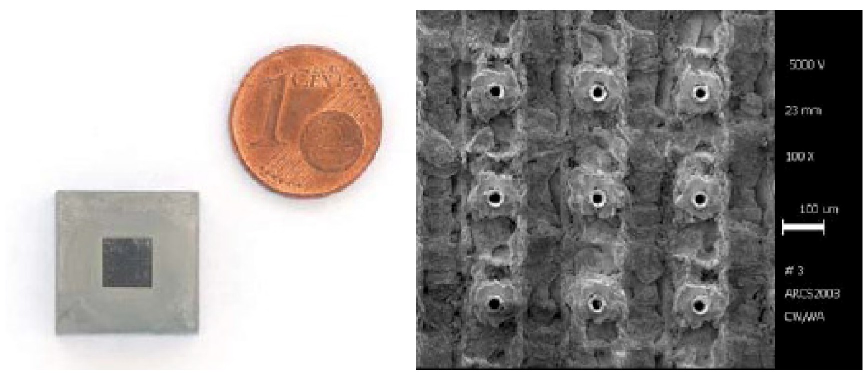


Fig. 38:Silica Bead Filled Silicon Micromachined Capillary Emitters by EFPL [113].

The Sandia/ARCS collaboration explored micro FEEP arrays using Indium as a propellant [114], [115]. Several different techniques to micromachined emitter arrays, including laser etching, micro Electric Discharge Machining (EDM), photochemical etching, and silicon micromachining were explored [114], [115]. Although the most precise machining results were achieved with silicon micromachined arrays, no coatings had been developed yet to allow indium to wet silicon. Hence, laser micro-etched arrays were used for initial demonstration tests (Fig. 39). Arrays consisting of 21 x 21 emitters, 90 μm in height with an I.D of 10 μm were machined and used in these tests [114], [115]. Currents of up to 4.5 μA were collected from the entire array. There was evidence that not all emitters fired at all times, which was attributed to the early development stage and related alignment issues, among others [115].

Microfabricated colloid arrays were also developed and tested and a quite early in the previous decade at Tsinghua University in Beijing, China [117], [118]. Tests with both silicon micromachined arrays [117] as well as a rather unique PCB board based array type [118] were conducted. In the case of the silicon based arrays, up to 192 emitters per array were fabricated (Fig. 40). Test units featured 4 emitters, and produced a measured thrust determined on a pendulum thrust stand of 6.8 μN. No information on the propellant was given [117]. An emitter voltage of 1480 V was required for onset, and voltage was driven to 1970 V to obtain the aforementioned thrust data [117].

A lower-cost approach was pursued by the Tsinghua team, replacing silicon micromachined arrays with PCB-board derived arrays [118]. Here, the copper deposited onto the PCB board is etched to form circular holes, connecting to the conducting liquid to apply the emitter voltage. A second PCB circuit board with larger diameter holes etched into the board served as extractor (Fig. 41) [118]. The array featured 81 emitters of 0.3 mm I.D. Thrust values of 2.6 μN were achieved with the array, measured on a



(a) (b)

Fig. 39 (a) Laser Micromachined Emitter Array by Sandia, (b) Emitter Array Chip [115]

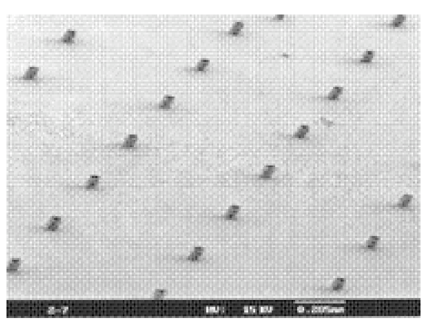


Fig. 40: Microfabricated Colloid Thruster Array of Tsinghua University, Beijing, China [117]

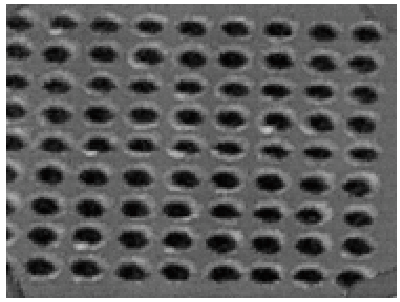


Fig. 41: PCB Board based Micro-Colloid Arrays by Tsinghua University, Beijing, China [118]

pendulum thrust stand. Once again no information reg. the propellant used was given. Tests were conducted in atmosphere, and voltage breakdown lead to limited thruster lifetimes [118]. A significant amount of effort was spend on the development of silicon-based microfabricated pump to supply propellant to the chip, with a flow rate capability from 10 – 300 μl/min. Pump mass was 20 g, chip dimensions were 15 x 15 x 4 mm, and the pump required 200 mW to operate.

Microfabricated electropray arrays offer significant potential for future microspacecraft applications, and beyond. They combine the inherent small physical dimensions of emitter tips in a compact, microfabricated package. Micro-colloid arrays offer high thrust-to-power ratios as a result of their lower specific impulse performance and offer dual mode operation (droplet and ion), significantly extending mission flexibility, yet typically require pressure fed systems. Ionic propellants used are also highly water absorbent, requiring careful propellant handling and feed system design to avoid bubble formation due to water evaporation. Micro-FEEP arrays, in addition to offering the same level of miniaturization as do colloid arrays, offer the system benefit of capillary propellant feed, allowing for extremely compact, low-complexity, no moving parts feed systems, significantly easing integration. However, at the present time, no dual mode operation has been demonstrated with FEEP systems yet, and currently conventional FEEP systems are constrained to high specific impulse operation at lower thrust-to-power ratios than colloids. Despite the miniaturization achievable for colloid arrays, and FEEP arrays with integrated feed systems, both will require miniaturized PPUs to meet the CubeSat design envelope, which will be an area of required future development. Depending on array size and thrust levels desired, power levels may also require deployable arrays. If dual mode operation is feasible, micro-electrospray arrays could span a significant range of propulsive applications, from delta-v (both low thrust/high Isp and high thrust/low Isp), precision pointing, and slew in a single package.

<u>Nanoparticle Field Extraction Thrusters (NanoFETs)</u>

A novel thruster concept, the nanoparticle Field Extraction Thruster (NanoFET) is an electrostatic thruster concepts that emulates electrospray thrusters, yet does not rely on droplet formation or ion emissions from a conductive liquid. Rather, it extracts solid particles of predetermined size from a liquid or powder reservoir [119]-[126]. Using particles of know size allows on to target specific impulses as needed for the mission, if multiple reservoirs with different particle sizes were onboard the spacecraft, the specific impulse could be tailored to the application through out the mission, i.e. from high thrust-to-power and low Isp for rapid maneuvers, to low-thrust-to power and high Isp for large delta-v applications.

Two different approaches are being explored at that institution. A liquid reservoir NanoFET relies on microparticles, either small conductive Al spheres or potentially nanotubes in later applications, suspended in a liquid with low vapor pressure, such as silicone oil [119], [122]. A micropump recirculates the liquid, transporting the particles past a charging electrode. An integrated extraction and accelerator grid system extracts the charged particles from the liquid reservoir and accelerates them to produce thrust. By emitting oppositely charged particles from different cells, no separate neutralizer would be required. The system was demonstrated with 1.5 mm long by 300 μm wide cylindrical aluminum particles, incl. in a reduced gravity flight experiment [125].

A dry powder particle extractor was designed to overcome several challenges identified with the liquid reservoir system, including maintaining a stable liquid/vacuum interface without fluid loss due to evaporation, fluid loss due to wetting of the particles, viscous drag in the liquid during extraction, and maintaining liquid purity to avoid electric breakdown [123]. In addition, fluid flow would need to be maintained to continually pass by the charging electrodes, requiring some sort of pumping mechanism, which could add complexity and would need to have adjustable pumping speeds or accept lower thrust levels with time as the reservoir is being increasingly depleted of particles.

Using a dry powder of ultrafine particles may lead to caking or clumping of particulates due to van der Waals forces. The dry NanoFET therefore relies on pushing a reservoir of particles through a sieve via piston action to separate the particles. Beyond the sieve, a grid system similar to the one in the liquid NanoFET is used to charge and accelerate the particles to produce thrust [123], [124]. The piston approach seems complex, and piezoelectric actuation of the sieve is envisioned to aid the extraction process [123]. Spherical particles ranging in size from 53 μm to 305 μm were used to demonstrate this concept [124].

No performance data have been obtained for either NanoFET type yet. Estimates range from under 200 sec Isp for solid 50-nm gold particles at up to 40 kV, up to 500 sec for solid ceramic particles of the same size, and up to 900 sec for hollow ceramic particles with a 5% by diameter shell thickness, which have yet to be demonstrated. Specific impulses are thus limited and accelerating voltages very high, which may make these concepts difficult to realize on Cubesats.

Other Nanoparticle Thruster Concepts

Two other nanoparticle thruster concepts have been proposed. The University of Kiel in Germany is exploring contact charging [127]. A needle emitter array is either coated or sprayed with nanoparticle, which acquire a charge upon contact and are emitted from the emitter needles and accelerated by a counter electrode. Obvious issues with this concept are propellant supply and feeding mechanisms.

At the University of Washington, a liquid colloid thruster concept is being studied that does not rely on capillary or needle emitters [128]. Instead a conducting liquid propellant sheath deposited onto a diaphragm is excited into a standing wave pattern via a piezoelectric actuator attached to the diaphragm. This principle is used today in commercially available humidifiers. However, in this concept the wave amplitudes are adjusted such that they remain just below the conditions for atomization. Standing wavelengths are on the order of microns [128]. An electric field is then applied to the standing wave, exerting an electrostatic pull near the wave crest, overcoming the surface tension forces to eject charge liquid droplets. Using salt water, the basic feasibility of this concept was proven, emitting a current of approx. 65 nA at 3 kV applied voltage [128]. An obvious issue with this concept is the liquid/vacuum interface in zero-gravity conditions.

Resistojets

Resistojets are among the simplest electric propulsion devices in concept. A liquid or gaseous propellant is passed through a heat exchanges, increasing its temperature, then expanded through a nozzle to produce thrust. Elevating the propellant temperature increases specific impulse and thrust. State-of-the art thruster hardware, however, is far too heavy and power consuming for microspacecraft, and in particular Cubesats [6]. However, a few newer developments deserve mention.

Ammonia resistojets are being developed by Busek in collaboration with GSFC (Fig. 42) [129]. The thruster weighs 15 grams without valving, delivers 150-210 second specific impulse at 5-12 mN of power and a power level of 25 W [129].

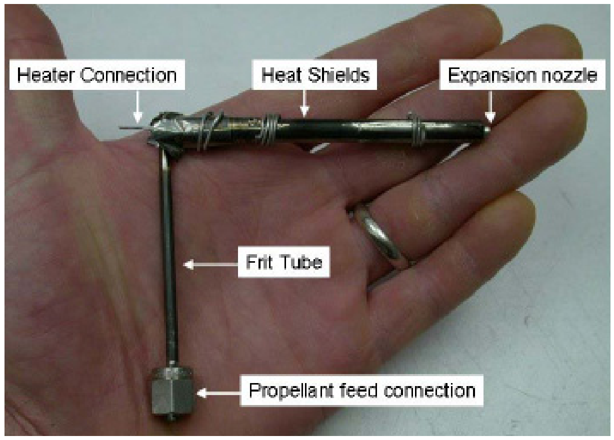


Fig. 42: Busek Ammonia Microresistojet [129]

A water resistojet has been under development at Surrey Space Centre in England, and was operated in space flight onboard an 88-kg microsatellite as part of the Disaster Monitoring Constellation (DMC) series of spacecraft (Fig. 43) [130]. Heater requirements were limited to 3 W to be compatible with future Cubesat requirements [130]. A single tank was used, without a plenum, and heater wire wrapped around downstream tubing to ensure complete vaporization. The propellant vapor would be further heated in the resistojet thruster itself. However, during testing ice formation was observed in the nozzle since the water vapor expanded in the nozzle and cooled. Additional heating or reducing the expansion ratio were considered two solutions. The latter was used. In-space demonstrations indicated that a lot more thrust was produced than anticipated (3.3 vs 0.2 mN), and in subsequent maneuvers it was found that the propellant tank was depleted. Apparently, liquid propellant has exited the thruster.

The icing issue in a MEMS-fabricated water resistojet was previously observed in experiments at JPL with a Vaporizing Liquid Micro-Thruster (VLM) (Fig. 44) [131]. The VLM used a maximum of 2 W of heater power in order to keep in compatible with future microspacecraft constraints. At these low power levels, however, the thruster was not able to operate stably, as flow rates and hence feed pressure had to be set so low that vapor pressure induced inside the thruster chip induced chugging. The aforementioned icing problem was a result of the same low power constraints. As a result of these issues and power constraints a butane thruster design by VACCO was adopted for the JPL Micro-Inspector spacecraft [8].

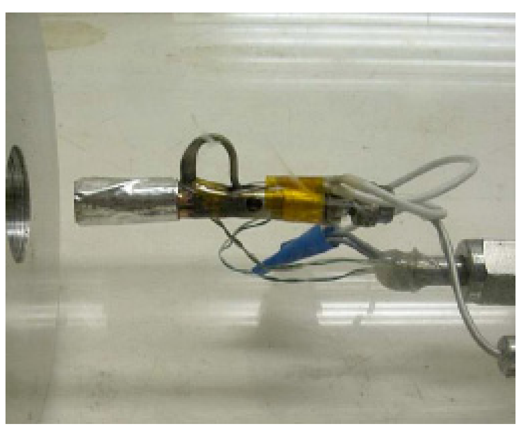


Fig. 43: Surrey Micro-Water Resistojet [130]

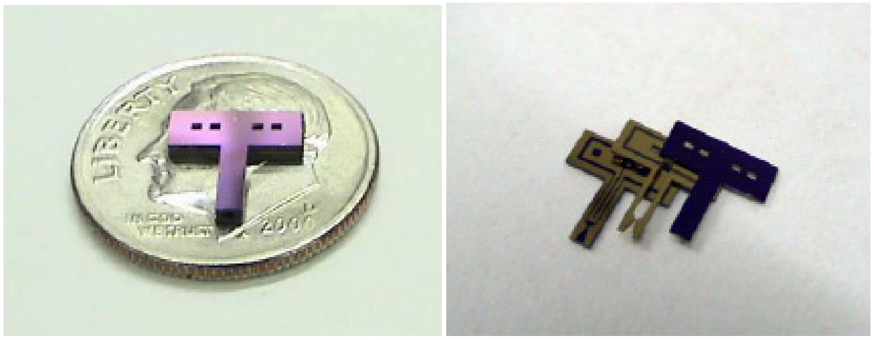
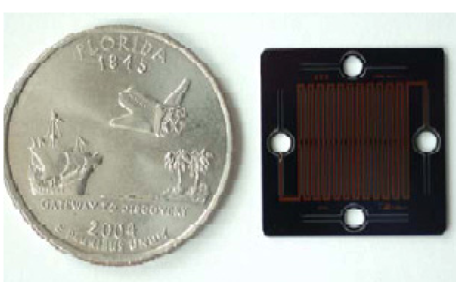


Fig. 44: JPL Vaporizing Liquid MicroThruster (VLM) [131]

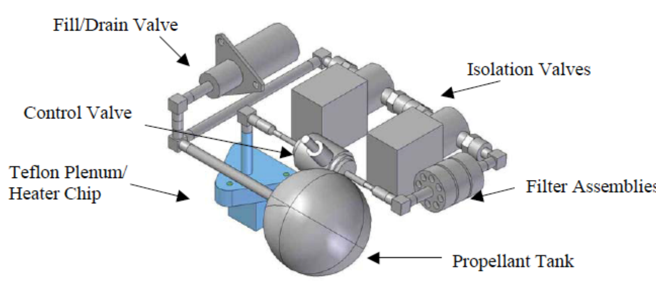
Ketsdever et al. has been exploring a novel type of Free Molecule Micro-Resistojet (FMMR) [132], [133]. In this concept, propellant s fed from a plenum as a very low pressurant gas such that the flow is in the free molecular regime, i.e. collisions over the characteristic thruster dimensions only take place with the hot wall surfaces, not among the molecules of the flow. Due to the lack of momentum exchange between flow particles, all particles colliding with the heater wall surfaces will have the kinetic energy equivalent to the heater wall temperature. Thruster geometry is chosen such that all particles entering the thruster must intersect the wall heater surface. [132]. Liquid propellant can be used if it is vaporized first when entering the plenum tank. Using water propellant, performances of 129 μN at 79 sec specific impulse were demonstrated at 3.2 W steady state power (5W maximum during heating transients) [133].

Figure 45 shows the FMMR heater chip, as well as the technology demonstrator propulsion system lay-out, which was constrained by commercially available valve hardware. This latter image demonstrate well the need for integrated feed system approaches, as alluded to in the introduction, a fact that Lee et al [133] are pointing out when referring to future, MEMS-based valve options that could be directly integrated with the FMMR chip, such as MEMS based piezovalves [134] or isolation valves, like the JPL Micro-Isolation Valve (MIV) [135].

Resistojets, despite their simplicity, offer only little performance improvements over cold gas systems at power levels, which, although small by conventional standards, may present a significant demand on a CubeSat power system. In the case of the aforementioned JPL Micro-Inspector satellite, the JPL VLM concept was therefore abandoned in favor of a newly developed butane thruster by VACCO [8]. Although lower in specific impulse performance than resitojet concepts, the power demands, other than during valve cycling, are significantly reduced over resistojet concepts.



(a)



(b)

Fig. 45: FMMR (a) Heater Chip and (b) System Layout [133]

Microcavity Discharge Thruster

The Microcavity Discharge Thruster (MCD) is a novel electrothermal thruster concept. It relies on flat panel microplasma heritage, realizing discharges in cavities as small as 10 μm in diameter at pressures up to well above 1 atm. Gas temperatures may reach 1500 K or higher at up to 2 W of power deposited per cavity, and if expanded through a nozzle, an electrothermal microthruster array concept can be realized [136]. Arrays with up to 1 million cavities have already been realized, allowing for significant scalability of this thruster concept, similar only to micro-electrospray arrays, potentially spanning power ranges from milli-Watt to kilo-Watt levels [136], yet operating in a higher thrust and lower specific impulse regime as the latter due to its electrothermal character.

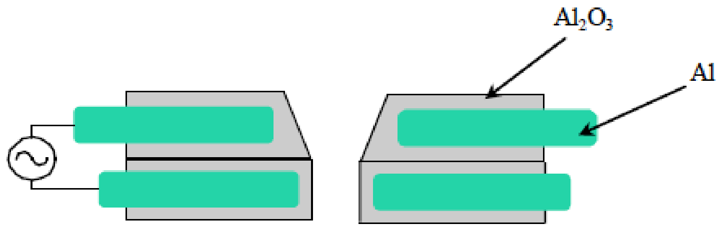
The thruster concept consists of two perforated aluminum foil electrodes onto which an aluminum oxide layer is grown. The two electrode sheaths are then bonded, and the perforations form the discharge cavities. A chemical etch forms a nozzle per cavity at one side of the electrode sheath. Applying a 50-150 kHz, 400-1200 VAC to the electrodes creates alternating electric fields inside the gas filled cavity, leading to low (<<1%) degree of ionization and subsequent heating of the partially ionized gas in the alternating electric field. Figure 46 shows a the thruster concept and etched nozzle [136]. Computational modeling of such discharges [136], [137] and initial thrust stand experiments have been performed with thruster prototypes at up to 0.25 W per cavity using neon with added nitrogen and water vapor to improve power absorption [136]. Thrust values of 0.6 – 2.7 mN were obtained at 120 – 240 kPa at 0.99 – 5.22 mg/s for a 4 cavity array with 120 μm throat and 210 μm exit plane diameter.

Microcavity discharge thruster arrays are an interesting thruster concept in early stages of development, for both microspacecraft applications and beyond. It extends the thrust range that could be delivered from electrospray arrays to higher thrust levels at higher thrust-to-power, yet lower specific impulse values. It operates at higher gas temperatures than conventional resistojets, promising higher specific impulses than the latter, and is highly scalable by appropriately sizing the array. Microcavity arrays may therefore fill a gap between higher Isp and lower thrust-to-power electrospray arrays and higher thrust, lower Isp chemical systems, and be used for attitude control, but also for rapid slew maneuvers and proximity operations, were relatively responsive spacecraft are required, as well as for low delta-v applications.

PROPELLANTLESS PROPULSION

Electromagnetic Formation Flying

Electromagnetic formation flying has been developed by for potential application to future interferometer missions for conventional spacecraft, and are currently under development by two research groups at MIT and in Japan [138], [139]. Since microspacecraft applications may involve swarms of these craft, in particular in Earth orbit where disturbance forces acting on a constellation may induce perturbations that need to be corrected against, electromagnetic formation flying may appear as a



(a)

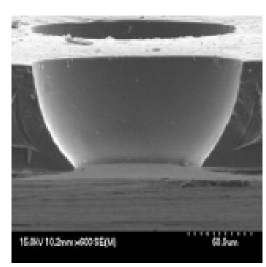

(b)

Fig. 46: (a) MCD Electrothermal Thruster Concept and (b) Micronozzle [136]

candidate for formation flying control as well. However, electromagnetic formation flying involves several limitations, which are amplified for microspacecraft, such as CubeSats.

In order to generate sufficient forces, high current electromagnetic coils must be used. For conventional spacecraft, the use of high-temperature superconducting coils is therefore considered, requiring to operate at temperatures below 77 k [138]. For conventional future interferometry missions, requiring cryocoolers for optical instruments, cryocooling of the electromagnetic coils is therefore not a significant added impediments. Microspacecraft, and in particular CubeSat platforms are unlikely candidates for sensitive optical instruments, and cryocooling would in all likelihood be an added requirements difficult to realize in such a small spacecraft envelope. Obviously, forces required to control a microspacecraft constellation are less than those for conventional craft, but as was seen in the Introduction, power levels available on CubeSats are equally constrained, shifting the problem to smaller, lower power electromagnets, but not eliminating it given the much more severe power constraints of CubeSats.

In addition to either attractive or repelling forces, two interacting electromagnets may induce torques when pointed orthogonally with respect to each other. These torques will cause the formation to rotate around its center of mass. Conservation of angular momentum will then induce spinning of the individual craft around their center of mass, with an angular momentum equal and opposite to that of the formation. This added complexity requires the use of momentum wheels to compensate. Theses wheel, however, may saturate over time, requiring propulsive means to desaturate them, magnetic torquers, or a scheme to sinusoidally vary the magnetic dipoles to result in a zero net gain of momentum of the wheels, thus avoiding wheel saturation [139].

Electromagnetic formation flying, although avoiding complexities related to the integration of a propulsion system, has its own complexities to deal with, including the management of spacecraft torques, cryocooling, limited range of force action for a given power level, and the fact that only interacting forces between spacecraft can be induced. Electromagnetic formation flying offers no delta-v, drag-make-up, or deorbit capability, and maneuverability is thus rather constrained. Attitude control, however, would be achievable via the reaction wheels required onboard.

Electrodynamic Tethers

While electromagnetic formation flying can only provide relative spacecraft motion and attitude control via reaction wheels, electrodynamic tethers have the opposite constraint. Tethers may provide de-orbit [140] or orbit raising capability [141] when used as a momentum exchange tether between two couples objects. Tethers do not allow for attitude control or relative motion, as may be required for inspection satellites. Orbit raising also requires coupled spacecraft, placing further constraints on the spacecraft and mission design.

Probably the most interesting application of tethers are their use for de-orbit, as this is an issue of growing concern with increasing number of space objects in near Earth orbit, and which would increase in number even more significantly with potentially significant numbers of difficult to detect microspacecraft added. An electrodynamic tether used for de-orbit relies on Lorentz forces action on the electrons in the conductive tether as it intersects Earth’s magnetic field lines along it orbital path. A voltage difference is thus set up between the ends of the tether. If the circuit is closed by contacting the ends of the tether through the ionosphere, by means of field emission cathodes, for example, a current flows through the tether that will induce a force by interaction with Earth’s magnetic field and perpendicular to it and the tether, causing it to decelerate, and the spacecraft connected to it along with it [140].

Electrodynamic tethers may thus serve as a de-orbiting device if no propulsive means are provided on the spacecraft. In case they are, adding more propellant to an existing tank may be a more mass efficient option than adding an additional tether system for this purpose, pending a trade involving delta-v required for deorbit, specific impulse of the existing propulsion system, required tether length and its deployment system. CubeSats, however, have been used in the Multi-Application Survivable Tether (MAST) experiment to deploy and diagnose a tether experiment in low earth orbit for micro-meteroid impact[142]. Thus, in this experiment CubeSats are serving the tether experiment as a diagnosis tool, rather than the tether serving the CubeSat.

Solar Sails

Solar sails have been proposed for space missions in the past [143], however, no missions have been realized yet. Solar sails rely on collecting reflecting sunlight, and propelling the spacecraft due to the transfer of momentum from the sun's photons to the sail. Solar pressure acting on sails is small, on the order of 4.7 x 10^{-6} N/m^2 in Earth's vicinity [144], thus requiring large sail areas to propel a conventional spacecraft. In order to not add significant amount of mass to the spacecraft, sails have to be extremely light-weight, which needs to include the deployment mechanism. Microspacecraft may offer another application for this technology. Given the much lower mass of microspacecraft, sail sizes can be reduced, easing fabrication and deployment.

Solar sail applications would typically be for delta-v, in particular for interplanetary travel. However, as was noted in the introduction, use of microspacecraft for interplanetary missions may not take the form of deploying individual microspacecraft as stand-alone spacecraft to its destination, for reasons of limitations with respect to instruments that could be carried along, especially due to constraints with respect to optical instruments (realizable aperture sizes on microspacecraft) as well as communication constraints due to limited space for high-gain antenna equipment. Rather, microspacecraft may be used for these and other reason as adjunct spacecraft to a larger mother ship that deploys microspacecraft for particular portions of the missions, such as those involving particular risks, for example. Therefore, this traditionally envisioned applications of solar sails to provide large delta-v along interplanetary trajectories may not be a major driver for microspacecraft.

A few niche applications have been identified, however. These may involve missions to study earths' geotail [143], [144], or an early magnetic storm warning network [144]. In the former, a constellation of microsats deployed throughout the geotail region allows this constellation to move along with the geotail , i.e. the Earth-Sun line, as it moves around the sun. In the latter solar sails allow a spacecraft to orbit the sun at a smaller radius than Earth's orbit, yet at an angular velocity equaling that of Earth, placing the spacecraft always between earth and the Sun. The difference in force between Sun's gravity and centrifugal forces due to orbiting at a lower angular velocity than required for this orbit is taken up by the solar sail. Placing a constellation of microspacecraft at a location between the Earth and the Sun allows for an early warning network against solar storms [144]. New technologies are being studied that may be applicable to microspacecraft, such as curved thin shells produced by Mevicon, Inc..[144]. These shells are stowed compactly, but deploy to their inherent stiffness without required actuation mechanisms or booms (Fig. 47) [144].

Except for these unique cases, near Earth applications of solar sails may take a rather different from. Rather than relaying on the reflectivity of the sail to produce thrust, sails may be deployed from CubeSats as drag chutes for deorbit (Fig. 48) [144]. This approach would compete with simpler drag devices, such as spring-loaded deployable rigid surface from the spacecraft, or inflatable balloons. It may be used in such applications, were large a large area is required for relatively rapid deorbit, warranting the added complexity of sail deployment. Once again, if a propulsion system is already onboard the spacecraft, the added cost, complexity, mass and risk of adding a separate de-orbit system would have to be traded against adding additional propellant to an existing thank for this purpose.

Fig. 47: Self-deploying Thin Film Shells by Mevicon [144]

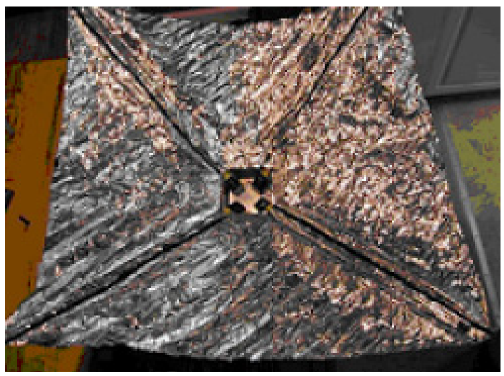


Fig. 48: Stanford Drag Sail for CubeSat Applications made from Mylar Survival Blanket Material [144]

Finally, as for tethers, CubeSats have been envisioned to be used as technology demonstrators for sub-scale solar sails, demonstrating deployment, solar sail dynamics or survival in the space environment. The CubeSat mission "NanoSail D" was to serve this purpose, but unfortunately did not make it to orbit due to a launch failure [145].

SUMMARY AND CONCLUSIONS

In order to increase the capability of future CubeSats, propulsion, notably absent on present day iterations, is required. Propulsion will allow for formation flying applications, orbit change (altitude and plane change), proximity operations, more precise attitude control and therefore significantly increase mission flexibility over present day CubeSats. There are a vast amount of candidate propulsion systems available for future Cubesat applications, either under significant development or currently in research. In fact, since this author conducted his last review of the field about a decade ago, over a hundred new references could be added to this database, and this list may not even be all including. A survey of micropropulsion options is also always a survey of the entire field of propulsion, as virtually each conventional propulsion technology has its counterpart in the micropropulsion field, to which have to be added several technologies that are unique to the field, such as vacuum arc thrusters, micro-cavity discharges, and electrosprays.

Given the vast scope of this review, limits had to be self-imposed to keep it manageable. Most notably, no microvalve and other flow component technologies could be included in this review. This by no means should leave an impression of lessened importance of these components. In the contrary – miniaturization of thruster components only will not enable micropropulsion systems for CubeSats or other microsats, unless miniaturization of these components is addressed as well. At the same time, novel integration schemes need to be developed. Conventional integration by means of welded tube joints would give away a significant amount of the gains made in component miniaturization to the required minimum distances between components to facilitate integration in a conventional manner, and be labor intensive and costly. Novel approach as MEMS-based, on-the-chip integration or ChEMS^{TM} technologies currently being pioneered by VACCO Industries, are promising approaches.

In the case of electric propulsion systems, the integration aspect has to involve the power and power conditioning systems. Current CubeSat designs offer too little power for many electric propulsion concepts, thus forfeiting enhanced mission flexibilities, i.e. use on formation flying applications or the possibility of significantly increased delta-v capability as may be required for orbit change. Deployable solar arrays, for example, are required and need to be developed in tandem to enable high-Isp electric propulsion systems onboard such spacecraft in order to fully capitalize on the benefits of electric propulsion systems. Power conditioning units, required to provide the electric propulsion systems with the required voltages currently are a major mass driver in already miniaturized systems, such as micro-PPTs, for example. Providing compact and lightweight units may likely dominate several already quite mature micro electric propulsion developments going forward.

Comparing this review with the one conducted a decade ago [6], several additional trends can be observed:

First, although a decade ago, micropropulsion was often, although by no means by everybody, viewed almost synonymously with MEMS, this trend underwent a significant reevaluation. While MEMS does offer the potential for significant miniaturization and very high precision, it also present the propulsion engineer with many new challenges. Among the former are the challenges to deal with new materials and their properties, not necessarily overlapping with the requirements of propulsion systems. The high degree of miniaturization enabled by MEMS may be more than what is required, or even possible for some thruster concepts. Reducing thruster or plasma chambers to too small a size increases wall losses, and may thus lead to efficiency losses. On the other hand, the chip-based integration of MEMS thrusters with valves and other components so far has only been realized in few cases, as the realization of reliable, high-force and leak-tight valves in MEMS poses even greater challenges than the thruster itself. Packaging MEMS thrusters individually and connecting them with conventional components, however, as pointed out above, does not realize the full miniaturization potential of this approach. At the same time, miniature metal machining techniques have matured and produced impressively small, and highly integrated systems.

There are a couple notable exceptions to this trend, however. Micro-electrospray arrays or future MCD arrays may greatly benefit, or even be enabled by microfabrication approaches on large scale arrays. While neither of these approaches necessarily has to rely on silicon microfabrication, it is clearly the basis for most of the electrospray array approaches currently being pursued. Assembling large numbers of emitters conventionally would be costly, likely not realizable at comparable packaging densities as obtainable with MEMS, and be less reliable due to fabrication tolerances. In these areas, microfabrication concepts may have found a truly unique role in realizing these concepts. In fact, the scalability of these devices to much larger sizes may allow them to be applied to much larger spacecraft and power levels than was possible with conventional architectures so far.

A second trend that can be observed over the last decade is the increased level of international as well as university activities in the field. While these were both present from the beginning, their relative weighting appears to have increased over US government and industry activities. This neither an avoidable nor an undesirable trend, and signifies the excitement that this technology has generated worldwide. For many smaller nations microspacecraft are also an affordable way to enter the space business. On the other hand, the number of activities in the US, especially in the government field have decreased. While in the defense area, several focused activities have been maintained and have achieved impressive results, such as in the electrospray field, overall the level of activity has been on the decline. This was an avoidable trend, and appears to have had its reasons in the demise of technology funds in recent years at important national institutions, such as NASA, a trend that only just now is in the process of reversal.

Where does all this leave the CubeSat designer? Concretely, three major areas of micropropulsion applicable to CubeSats and similar sized microspacecraft can be identified: (1) Existing or Nearly-off-the-shelf technologies, (2) New technologies under significant past development that may enable significant increases in propulsive capability, and (3) Emerging research and development that allows an even greater level of integration and compactness, while affording far reaching propulsive capabilities for the future. Theses three areas will be briefly elaborated upon in the following.

(1) Existing or Nearly Off-the-Shelf Propulsion Technologies:

The following technologies are all nearly available for use on CubeSats today with only minor or no modifications required. However, they currently offer only limited propulsive capability. They include:

- **Butane Systems**: combining liquid propellant storage and low-mass tankage with the simplicity of a cold gas system, at high levels of integration using VACCO ChEMS^{TM} approaches, yet at limited specific impulse and delta-v capability. Sufficient for proximity operations. Using nitrous oxide, tank masses increase, but so will delta-v capability. Systems developed for MEPSI are ready for CubeSat usage.
- **Micro-Pulsed Plasma Thrusters (PPTs)**: based on the earlier AFRL μPPT development, integrated systems have been developed by Busek that have flown on 100-kg class microsats, and could be integrated onto CubeSats. Challenges remain PPU miniaturization, and the fact that total impulses of these systems are limited and that operation is inherently in a pulsed mode. Suitable for precision pointing and small delta-v.
- **Vacuum Arc Thrusters (VATs):** Systems have been developed specifically for CubeSat applications. As for PPTs, however, operation is inherently pulsed and total impulse is limited. Suitable for precision pointing.

(2) New Technologies under Past Significant Development:

These technologies have seen significant development, but require additional effort in subsystem miniaturization, such as feed system and power processing. If realized, they will significantly extend propulsive capabilities of future Cubesats. They include:

- **Monopropellant Systems**: Miniature hydrazine thrusters would offer significantly increased specific impulse and delta-v capability, yet provide thrust levels required for rapid orbit transfer. If feedsystem (valve) technologies can be optimized, lower power operation than presently available may be possible, which at this point prohibits their use on CubeSats. Hydrazine use onboard CubeSats has been viewed with skepticism in the past due to safety concerns, however, with the interest by government and industry into CubeSats this may no longer be a major impediment in those cases. Hydrogen peroxide use has been proposed, however, long term storability needs to be verified.
- **Ion Propulsion Systems:** Ion propulsion offers significant increases in Isp. Both DC as well as RF micro-ion thrusters have already been demonstrated, micro-feed systems and PPUs, however, remain to be developed. Also, to be applicable to CubeSat, their power capability needs to be increased significantly to greater than 50 W, likely through the use of deployable arrays. CubeSats of the "3U" architecture could be envisioned with a propulsion module featuring thruster, propellant tank and integrated feed system, a power module, featuring PPU and deployable arrays, and a payload/avionics/telecom module. Significant miniaturization has also been achieved recently for Hall thrusters at much improved efficiency levels, yet thruster masses are high due to magnetic circuitry.
- **Colloid Thrusters:** Leveraging significant development in colloid thruster system for New Millennium ST-7, single thruster heads of the ST-7 thruster could be adapted to CubeSats to provide delta-v capability. The high thrust-to-power ratios of colloid thruster at a lower Isp than ion systems fills a gap in performances between ion and chemical systems that may be of interest in particular to Earth-observing missions.

(3) Emerging Technologies:

- **Microfabricated Electrospray Arrays:** Microfabrciated Electrospray arrays were the first MEMS thruster concepts being proposed in the early 1990s, however, have emerged as working laboratory devices only over the course of the last decade. Electrospray arrays offer significant advantages with respect to compactness, in particular for liquid metal FEEP arrays not requiring pressure-fed feed systems, inherent modularity in those cases, and resulting ease of integration, as well as a large specific impulse range if dual mode operation (droplet and ion emission modes) can be adapted. The latter have already been demonstrated for colloid systems. The theoretically feasible scalability to very large arrays using microfabrication techniques may actually feed back into applications on conventional (non-micro) spacecraft. Electrospray arrays may perform attitude control, incl. precision pointing, as well as delta-v applications.
- **Micro-Cavity Discharge Thrusters**: Using microfabricated cavity discharges arranged into arrays, they are the electrothermal equivalent of electrospray arrays, yet operating at lower specific impulses and higher thrust levels than the latter, and are thus applicable for attitude control and delta-v applications with faster transfer requirements. Since using gaseous propellants, they do require feed system components that would need to be miniaturized.

Propellantless propulsion concepts, although repeatedly garnering increased attention, only have limited application to CubeSats and similar sized microsats. They either are unlikely to be used on microspacecraft, such as electromagnetic formation flying, due to high-temperature superconducting electromagnetic coils and associated cooling required, or their limited flexibility, either for specific niche applications, such as solar sails for Geotail or geomagnetic storm warning networks, or de-orbit. In the case of the latter, such system may make sense if no propulsion system is available on Cubesat or microspacecraft. Higher capable craft that feature propulsion systems may not benefit from adding a separate system for such specific applications when additional propellant can be loaded in already existing propellant tanks.

Adding propulsion systems to CubeSats will add complexity and cost in exchange for additional capability. In particular the large non-recurring development costs are beyond the capabilities of the typical university user. The recent interest by industry and government into CubeSats, however, offer an opportunity to contribute to the field with such developments and subsequently flying them, thus reducing future costs of these systems and making them more widely available. Thus, CubeSats may not only have a role in future government and industry space activities, adding new capability routed in their small size and large potential numbers in the form of spacecraft formations, but, vice versa, government and industry may yet have a role in CubeSat development by increasing the capabilities of these unique space assets.

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